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CONFERENCE ON SUPERSONIC MISSILE PROPULSION

A COMPILATION OF THE PAPERS PRESENTED

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

[redacted]

March 13, 1952

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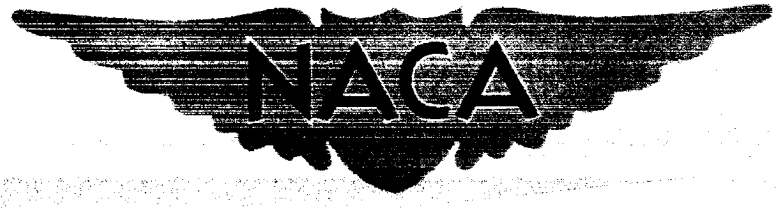

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ON

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INTRODUCTION

This volume contains copies of the technical papers presented at the NACA conference on Supersonic Missile Propulsion on March 13, 1952 at the Lewis Flight Propulsion Laboratory. A list of the conferees, who are members of the aircraft industry and the military services, is included.

The original presentation and this record are considered supplementary to, rather than substitutes for, the Committee's system of complete and formal reports.

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PREPRINT

of Figures for Talks presented at

NACA CONFERENCE ON SUPERSONIC MISSILE PROPULSION

LEWIS FLIGHT PROPULSION LABORATORY

March 13, 1952

I - MORNING SESSION, 9:45 A.M.

I - ENGINE CHARACTERISTICS

1. INTRODUCTION AND COMMENTS Ia
By Eugene J. Manganiello

2. ANALYSIS OF MISSILE PROPULSION SYSTEMS
By Eldon W. Hall, Hugh M. Henneberry and Philip J. Evans, Jr.

3. COMMENTS Ib

4. SOME RAMJET ENGINE PROBLEMS
By Roland Breitwieser, Eugene Perchonok and Thomas B. Shillito

5. COMMENTS Ic

6. THE TURBOJET ENGINE FOR MISSILE PROPULSION
By Bruce T. Lundin and Robert O. Bullock

7. COMMENTS Id

8. STATUS OF THE LIQUID PROPELLANT ROCKET ENGINE
By Gerald Morrell and Vearl N. Huff

[redacted]

II - AFTERNOON SESSION, 2:15 P.M.

II - ENGINE INSTALLATION

9. INTRODUCTION AND COMMENTS IIa
By DeMarquis D. Wyatt

10. SUPERSONIC INLETS AT HIGH MACH NUMBERS
By Edgar M. Cortright, Jr. and James F. Connors

11. COMMENTS IIb

12. VARIABLE INLETS AND OUTLETS FOR OFF-DESIGN OPERATION
By David S. Gabriel, Roger W. Luidens and Martin J. Saari

13. COMMENTS IIc

14. CONTROLS FOR MISSILE PROPULSION SYSTEMS
By Seymour C. Himmel

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LIST OF CONFEREES

The following conferees were registered at the NACA conference on Supersonic Missile Propulsion, Lewis Flight Propulsion Laboratory, Cleveland, Ohio, March 13, 1952:

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1. INTRODUCTION AND COMMENTS Ia

By Eugene J. Manganiello

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1. - INTRODUCTION AND COMMENTS Ia

By E. J. Manganiello

The material presented in the subsequent papers falls into two general subject categories. The first category, which is discussed in the first four papers, includes the performance, problems, and improvements of the propulsion systems proper and of their components. The second category, which is discussed in the last three papers, involves the general problems associated with installation of the power plant in the airframe, that is, the problems of the air inlet, the exit nozzle, and the engine control.

The engine types to be discussed are the ram jet, the turbojet, and the rocket. Before the review of the research investigations pertaining to these engine types is presented, the results of the analytical study made to determine, for each engine type, the desirable or optimum design characteristics and operating conditions for supersonic missile propulsion will be summarized.

The suitability of any missile propulsion system is, of course, highly dependent on the characteristics of the missile in which it is installed and on its flight plan or tactical mission. Any realistic analysis of missile power plants must therefore be properly and completely integrated both with the missile configuration and with its particular flight plan if significant results are to be obtained.

The different missiles and tactical missions of current interest are large in number and variety, and detailed and complete consideration of all of them is far beyond the scope of this report. Consideration can, however, be given to several broad classes of missile. After some preliminary study several representative flight plans and missile configurations were selected for detailed analysis. Photographs of models of some of these configurations are shown in figure 1.

These configurations are not presented as definitive or actual examples of specific missile designs; rather they are merely intended to illustrate general classes of configurations. No special merit is claimed for their aerodynamic shapes and arrangements - indeed, alternate arrangements in considerable variety and of equal or perhaps superior merit could have been selected. Care was taken, however, to incorporate in each of the configurations realistic aerodynamic structural and propulsive design principles. The configurations are thus considered fairly representative of current practice and thinking and should provide a useful and realistic evaluation of the relative importance and criticalness of the various propulsion system variables.

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In the presentation principal emphasis has been placed upon the relatively long-range missile problem, that is, ranges from 3000 to 6000 miles. Shorter ranges are, of course, of considerable interest and importance; however, analyses show that for short ranges missile performance is not particularly sensitive to engine type or engine design variables. In fact, general analyses of the effect of power plant characteristics on short-range missile performance are not very useful or conclusive and selection of the propulsion system is predicated much more upon logistics, engine availability, and other practical considerations than upon engine performance.

The analytical studies of the effect of engine type and design characteristics on the performance of long-range missiles are summarized in the next paper.

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Figure 1

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2. ANALYSIS OF MISSILE PROPULSION SYSTEMS

By Eldon W. Hall, Hugh M. Henneberry

and Philip J. Evans, Jr.

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2. - ANALYSIS OF MISSILE PROPULSION SYSTEMS

By Eldon W. Hall, Hugh M. Henneberry, and Philip J. Evans, Jr.

INTRODUCTION

The relations between engine performance and long-range missile performance are examined in this analysis in order to indicate the problems upon which research effort should be concentrated.

The engines included in the analysis were the ram jet, the turbojet, and the rocket. Boosting problems were considered and in general the results of arbitrary variations in engine performance parameters are presented in terms of their effect upon missile-plus-booster gross weight.

The effects of design changes in the missile configurations were not considered, but every attempt was made to incorporate consistent missile designs in analyzing the engine variables.

ANALYSIS

In calculations of the effect of engine performance on missile weight, many assumptions are necessary. Some of these assumptions describe the external aerodynamics of the missile configuration. Others relate the weight and structural requirements of the various missile components. Wherever possible, the latest available experimental data were used. No detailed discussion of the many analytical assumptions is included in this paper. Principal emphasis was directed toward incorporating consistent assumptions so that comparisons and trends could be evaluated.

This analysis was directed toward long-range missile requirements, and in general considered ranges between 3000 and 6000 miles. It appears that solutions to the problems of long-range supersonic missile design will not be forthcoming unless the most advanced techniques are applied. The assumptions and results of this analysis are therefore intended to represent advanced structural and engineering practices.

Structural weights were estimated assuming titanium was available wherever benefit could be derived from its use. For all calculations, a 5000-pound payload plus 2000 pounds of guidance equipment were carried in the missile fuselage. Fuel and control volume were provided in the fuselage; no fuel storage volume was assumed in the wing.

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In general, gross missile-plus-booster weight for a specified range has been used herein as a significant variable in evaluating the missile. It is often preferable to evaluate missile performance on the basis of structural weight, that is, the gross weight minus fuel and pay-load weight. The significant trends exhibited by the results of this analysis would be the same even if structural weight were used as the dependent variable.

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RESULTS AND DISCUSSION

Ram Jet

A typical long-range missile configuration powered by ram-jet engines is illustrated in figure 1. The design has a triangular wing of biconvex section mounted above the fuselage and the engines are mounted in nacelles beneath the wing. Wing loading is 75 pounds per square foot and the wing has an aspect ratio of 3, which results in a sweep angle of 53 degrees. Thickness-chord ratio is 3 percent. The fuselage is a body of revolution with parabolic silhouette and fineness ratio of 12. The vertical tail surface is 15 percent of the wing area and the horizontal tail surface, which has zero lift at design point, is 10 percent of the wing area. The missile is designed to fly at 70,000 feet altitude and a Mach number of 3.5. The entire missile is boosted by two rocket boosters attached beneath each wing, between the nacelles and fuselage. The boosters are illustrated along with the missile in figure 2, the upper pair representing the size required for full boost to design altitude and Mach number, and the lower pair representing the size required for boost to an altitude of 10,000 feet and a Mach number of 1.5. Gasoline-oxygen propellant having a specific impulse of 240 seconds was assumed for the boosters.

The boost path for the missile-booster combination is shown in figure 3. The missile is launched vertically, and follows a zero lift path to the design Mach number and altitude. At the design point the booster is dropped and the missile begins the cruise phase of the flight along a Breguet path. There are, of course, other methods of launching and boosting these missiles, but for full boost to design altitude and Mach number, tremendous booster sizes will be required as illustrated in the upper boosters of figure 2. In fact, the gross weight of the two large booster units is greater than the gross weight of the missile.

One of the primary factors which determines the size of the missile is, of course, the range. The effect of range on weight for the configuration considered is shown in figure 4, which presents missile plus booster gross weight as a function of range. For a range

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of 3000 miles, the missile plus booster weight is 45,000 pounds. As the range increases, the weight increases very rapidly until a missile-plus-booster weight of 150,000 pounds is required for a range of 6000 miles.

Figure 4 was computed for a cruise Mach number of 3.5 and an initial altitude of 70,000 feet. Missile performance at other Mach numbers and altitudes is presented in figure 5. As flight conditions are changed, it is necessary, of course, to change the configuration to retain the best aerodynamic and structural qualities of the engine-missile combination. For example, wing loading is changed to provide the best compromise between wing weight and lift-drag ratio.

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Figure 5 demonstrates the insensitivity of missile weight to changes in Mach number between 3 and 3.5 if the optimum altitude is selected for each Mach number. It is also evident that if design changes are made in the missile, the altitude can be varied considerably to cover a large range of ambient pressures. At a Mach number of 3.5, for example, the altitude could be varied between 60,000 and 75,000 feet without seriously affecting the weight. Even larger variations could be tolerated if such changes provided other improvements in performance, for example, increases in combustion efficiency.

The effects of larger changes in flight Mach number are presented in figure 6, which shows missile-plus-booster gross weight for Mach numbers between 2.5 and 4. Performance at each Mach number is for the optimum altitude in each case. Although the weights are high at low Mach numbers, less than 3, variations in Mach number between 3 and 4 result in only 10-percent variations in weight.

It appears, therefore, that rather large changes can be made in flight conditions without seriously affecting the weight, provided corresponding changes are made in the design of the missile.

When the large booster sizes required for the missile of figure 1 are recalled, it appears that the booster should be carefully analyzed in any attempt to reduce gross weight. One method of reducing the rocket booster size is to use the ram-jet engine for boosting the missile from some low Mach number and altitude to the cruise condition. It was found that if the geometry of the ram-jet engine were made continuously variable to provide good performance over a wide range of Mach numbers, it could easily boost the missile from Mach numbers much lower than the cruise Mach number.

A typical flight path for a boost of this type is shown in figure 7. In this case the rocket is used to boost the missile to Mach number 1.5 at low altitude, where the rocket is dropped and the ram-jet engine takes over. The missile then climbs and accelerates

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to the cruise Mach number and altitude with the ram-jet power and then continues on the cruise path to the final target. During the ram-jet boost period, the inlet diffuser, combustor, and exhaust nozzle are all required to operate over a wide range of conditions at or near peak efficiency.

Ram-jet boosting performance is improved as larger engines are used, and larger engines result if the cruising fuel-air ratio is lean. In this way, lean cruising operation reacts favorably on the engine boost characteristics.

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The effect of using the ram-jet engine during boost on the weight of the missile and booster is shown in figure 8. Weight is plotted as a function of ram-jet take-over Mach number. The take-over Mach number is defined as the Mach number at which the rocket booster is dropped and the ram-jet engine takes over; take-over altitude was varied along with take-over Mach number. The lower curve shows the weight of the missile at the take-over point. As the take-over Mach number is reduced more fuel is required by the ram-jet engine to boost the missile to the cruise condition and therefore the missile weight increases. On the other hand, the booster weight relative to the missile weight is reduced as the take-over Mach number is reduced. The upper curve shows the sum of the missile plus booster weights. For this configuration having a cruise Mach number of 3.5 and continuously variable engine geometry, the minimum missile-plus-booster weight occurred at take-over Mach numbers between one and two. Booster weight for boost to Mach number 1.5 is less than half the booster weight required for full boost to Mach number 3.5, resulting in a 25 percent decrease in total missile-plus-booster weight. The booster size required for boost to a Mach number of 1.5 and an altitude of 10,000 feet is illustrated by the lower pair of boosters in figure 2. Most of the weight saving is a result of the decreased weight of rocket propellant which is often relatively inexpensive compared to the cost of the remaining structure. It should be emphasized that the geometry for this engine was assumed to be continuously variable. Additional calculations showed that the use of a practical geometry arrangement for the ram-jet engine, such as two-position inlets and outlets, would reduce the saving in total weight to about one-half that indicated in figure 8.

Another way to reduce the rocket-booster size, without the added complication of requiring the ram-jet engine to operate over a wide range of Mach numbers, is to use in the rocket booster a propellant that has a higher specific impulse. The booster propellant used in constructing the previous figures was gasoline-oxygen and was assumed to have an effective specific impulse of 240 seconds. The effect of changing to a higher-impulse propellant is shown in figure 9. The

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upper and lower curves are the same as those in the previous figure. The dotted curve shows the total missile-plus-booster weight for a specific impulse of 325 seconds as compared to 240 in the upper solid curve. This impulse could be obtained with several propellant combinations currently under study. At any take-over Mach number, the size of the booster is reduced by the improvement in rocket propellant. At the high take-over Mach numbers, the savings are largest because the booster weights are largest. With the higher-specific-impulse rocket booster, the maximum reduction which could be realized with a continuously variable ram-jet engine is shown to be about 16 percent.

2546-8

The previous two figures have indicated that worth-while weight reductions can be obtained with continuously variable ram-jet engine geometry. Two-position engine geometry reduces these savings by about one-half, and even two-position geometry represents serious design complication as compared with fixed engine geometry. In view of these arguments, full rocket boost to design altitude and Mach number together with fixed geometry ram-jet engines still appears to be a practical compromise. Accordingly, the remaining ram-jet discussions will be based on this boosting technique.

Inasmuch as the booster weight is dependent on the missile size, any reduction in missile weight will have a two-fold effect in reducing the total weight of missile plus booster. One feasible method of reducing the missile weight is the use of drop fuel tanks during cruise. A missile of this type is illustrated in figure 10. In this configuration, a single ram-jet engine is located in the lower part of the fuselage and is provided with a bottom inlet. Fuel is carried both in the fuselage and in the external wing-tip tanks. The external fuel tanks along with parts of the wing can be dropped during cruise, and if a large part of the fuel is carried in the detachable tanks, large weight and drag reductions can be realized by dropping them as soon as their fuel is exhausted. In order to avoid excessive increases in the altitude which should accompany the reduced weight and might seriously affect the combustor performance, parts of the wing are dropped along with the fuel tanks. The flight path for this configuration is illustrated in figure 11. The missile is boosted to cruise Mach number and altitude with a rocket booster. The booster is then dropped and the ram-jet engine uses fuel from the external tanks. When the missile has flown within 3000 miles of its destination along a Breguet path, the external fuel tanks are empty and they are dropped along with parts of the wing. With a small change in engine fuel-air ratio and descent to the initial cruise altitude, the missile continues at the same Mach number along a Breguet path until the target is reached.

Reductions in weight and drag are, of course, realized when the wing tips and tanks are dropped, and these reductions are reflected in

[redacted]

lower total fuel load and initially smaller missile size as can be seen from figure 12. In this figure is shown the gross weight of missile plus booster as a function of range for the original configuration and for the configuration with detachable tanks and wing tips. With dropped tanks and wing tips, as would be expected, the benefits of staging are most pronounced at long ranges where the savings in tank weight and drag are greatest. At 6000 miles range, the missile-plus-booster weight is reduced 25 percent. It is obvious, however, that this saving is made only at the expense of added complication.

2546-9

Schemes have been considered which can reduce missile weight without a change in ram-jet engine performance. The remaining ram-jet discussion will be centered on the engine itself in order to determine what characteristics and design parameters are important.

Because the fuel weight is a large part of the missile weight, any engine variable which tends to reduce the fuel consumption is of first-order importance.

One of the primary factors in the ram-jet engine which determines the quantity of fuel used is the combustion efficiency. In the previous curves a combustion efficiency of 85 percent was assumed. The effect of combustion efficiency on missile-plus-booster weight is shown in figure 13, which includes efficiencies of 0.75, 0.85, and 0.95. This efficiency is seen to be very important even at the shorter ranges. As the range increases, the effect of efficiency increases and soon is a primary consideration in determining whether the missile can be designed to travel the required distance.

Resort to high-energy fuels has much the same effect as an increase in combustion efficiency. A high-energy fuel is considered to be a fuel having a high heat content per unit weight. The effect of changing to a typical high-energy fuel, pentaborane, is shown in figure 14. The missile-plus-booster weight is shown as a function of range for a typical hydrocarbon fuel and for pentaborane. It is immediately evident that large savings can be made on a weight basis. At 6000 miles the weight can be reduced to about one-half the weight with a hydrocarbon fuel. There are many other factors associated with the use of high-energy fuels, however, such as availability, handling, and storage, all of which must be considered.

Of somewhat lesser importance than combustion efficiency is the engine weight, the effect of which is shown by figure 15. The engine weight is not of primary importance because it is only a small part of the total missile weight. As with the other parameters, however, the effect of changing engine weight increases as the range increases.

[redacted]

In all other calculations the engine weight was taken as 180 pounds per square foot of combustion-chamber area. Curves are also shown for 120 and 270 pounds per square foot, representing more than a 2-to-1 change in engine specific weight.

One engine component which affects both engine thrust and fuel consumption is the exhaust nozzle. The effect of exhaust-nozzle velocity coefficient is shown in figure 16. The nozzle velocity coefficient is of primary importance because it affects the jet thrust directly and, to a much larger extent, the net engine thrust when the inlet momentum and engine drag are subtracted. At a Mach number of 3.5 (fig. 16), the weight of a typical missile plus booster for long range may be increased as much as 35 percent by only a 2.5-percent decrease in velocity coefficient. Even at lower Mach numbers, the effect is equally important because of the lower engine-cycle efficiency.

2546-S

Another engine component, the inlet diffuser, is also important because of its effect on engine air flow, cycle efficiency, and engine drag. Further discussion of the inlet diffuser is included in other papers in this conference.

The results of the ram-jet analysis may be summarized as follows: The primary ram-jet engine factor influencing ram-jet missile-plus-booster weight is the efficiency of the components, whereas engine weight may be considered of secondary importance. It was also found that all the engine variables become increasingly important as the range increases. Reductions in the weight of the booster could be obtained by the development of high-specific-impulse rocket propellants.

Turbojet

Consideration will now be given to the characteristics of turbojet engines that are desirable for supersonic missile propulsion. Because the turbojet engine is not dependent on high flight speeds to develop thrust, it is much better suited than the ram-jet engine to replace part of the rocket during boost. Although current turbojet engines are not built to withstand the temperatures encountered at all Mach numbers considered for the ram-jet engine, its speed range could be extended considerably with some modifications. In this analysis Mach numbers up to 2.5 were considered for the turbojet engine.

A typical turbojet powered missile is illustrated in figure 17. The small booster located underneath the fuselage is used to boost the missile to a Mach number of 0.4. The booster nozzle is oriented so that the rocket motor thrust is directed through the center of gravity of the missile-booster combination. Aerodynamically and structurally the missile is similar to the ram-jet configuration discussed previously.

[redacted]

A typical boost-flight schedule for the turbojet missile is presented in figure 18. The rocket booster accelerates the missile to 0.4 Mach number at low altitude and then separates. By means of the turbojet engines, the missile then accelerates to a Mach number of 0.8, climbs subsonically to 30,000 feet, accelerates to supersonic speeds, and then climbs and accelerates to the design Mach number of 2.5 and altitude of 55,000 feet. For efficient engine operation during the boost period, it was assumed that the engines were equipped with continuously variable inlets and outlets. In contrast to the ram-jet engine, all turbojet performance is presented for a boost of this type, utilizing a small rocket booster and the turbojet cruise engines. Increases in the range required large increases in weight as shown in figure 19, in which missile-plus-booster weight is presented as a function of range. The weight required for a range of 3000 miles is about 35,000 pounds and for longer ranges, the required weight increases rapidly. In fact, the weight is doubled when the range is increased from 5000 to 6000 miles.

2546-B

The performance of figure 19 was obtained for a Mach number of 2.5. The effects of lower cruise Mach numbers are presented in figure 20, for a range of 3000 miles. Missile-plus-booster weight is not sensitive to changes in Mach number between 2 and 2.5 and a change from 1.5 to 2.5 results in a weight decrease of only 15 percent. At each Mach number, of course, the optimum altitude and missile design were selected to give the best compromise between weight and efficiency of the engine-missile combination.

Although no large reductions in weight could be realized by changing the Mach number between 1.5 and 2.5, flight at subsonic speeds where higher lift-drag ratios are possible would result in much lower weights for the same range. In order to avoid the tactical disadvantage of subsonic flight, however, an interesting configuration that will be considered flies subsonically during the last phase. Such a missile together with its booster is illustrated in figure 21. The configuration consists of a supersonic stage attached underneath a subsonic stage. The turbojet engines are attached to the supersonic stage and are used for both subsonic and supersonic phases. The supersonic stage has a delta wing of biconvex section with a sweep angle of 63 degrees. The vertical tail surface is inverted and the high-sweep delta wing obviates a horizontal tail surface. The subsonic stage represents conventional high-speed subsonic design. After boosting, the missile flies subsonically until fuel is exhausted from the fuselage of the subsonic stage. The missile then separates, discarding the subsonic fuselage and wings but retaining the engines on the supersonic portion. The supersonic portion, which also carries the 5000-pound pay load, begins the final supersonic cruise of 3000 miles with a full load of fuel.

[redacted]

The flight schedule for the two-stage, two-speed turbojet missile is presented in figure 22. The missile is boosted to 0.4 Mach number at sea level with the rocket booster where the turbojet engines take over. The missile then climbs and accelerates to a Mach number of 0.8 and altitude of 35,000 feet where the subsonic cruise is started. After separation, the second stage accelerates, climbs to an altitude of 55,000 feet, and cruises to the target at a Mach number of 2.5.

2546-G

The advantages to be gained from the staged configuration are shown by figure 23. Missile-plus-booster weight is plotted against range for the original configuration of figure 19, and for the staged configuration. The advantages of the staged configuration are especially large at long ranges. At 6000 miles, for example, the missile-plus-booster weight may be reduced to as little as 35 percent of the weight of the single-stage all-supersonic configuration. This, of course, is at the expense of much greater complication and is due largely to flying at subsonic speeds during part of the flight.

Although large gains are possible for long ranges by staging, the effects of engine-design variables on missile performance will be studied for the conventional single-stage supersonic missile. The effects of the engine variables on the 3000-mile missile could be applied to the second or supersonic stage of the staged configuration.

The effect of compressor pressure ratio is shown in figure 24. The pressure ratios shown are those which would result with rated sea-level operation of the engine. Because of higher compressor-inlet-air total temperature, the pressure ratios at the cruise condition are much lower. For the 3000-mile-range configuration the weight is unaffected by changes in rated pressure ratio between 3 and 7. For the long-range configuration of 6000 miles, low weight was obtained with rated pressure ratios of 3 to 5. In these calculations, no account was taken of a change in engine weight per unit frontal area with a change in pressure ratio. Of course, any decrease in engine weight with decrease in pressure ratio would shift the optimum pressure ratio to a lower value.

The effects of engine weight are shown in figure 25. The engine weight is presented in terms of the specific weight that the engine would have without an afterburner at rated sea-level conditions. Unlike the weight of the ram-jet engine, which may be considered of secondary importance, the weight of the turbojet engine is of primary importance, especially at the longer ranges. The reason the weight is so much more important for the turbojet engine is that it is a much larger part of the missile weight and replaces a large amount of fuel which could be carried to increase the range. Of course, one way to reduce the engine weight is to use higher afterburner temperatures. As the afterburner

temperature increases, the required engine size decreases. On the other hand, the fuel consumption increases with the net result that these factors very nearly compensate and satisfactory missile performance can be obtained with a wide range of afterburner temperatures. Optimum afterburning temperatures would probably be determined from the requirements of the engine during boost and from practical considerations of burner weight, drag, and efficiency. Discussion of these factors is included in other papers in this conference.

2546-B

From engine-cycle considerations, it appears that increases in the turbine-inlet temperature will provide large increases in engine thrust and reductions in size with smaller penalties in fuel consumption than result from operation with an afterburner. The effect of using a higher turbine-inlet temperature is shown in figure 26. Curves are shown for the turbine-inlet temperature of 2000° R used in all other calculations and for a higher temperature of 2500° R. For the 2500° curve, 5 percent of the engine air flow was used to cool the turbine blades. For the longer ranges, increasing the turbine-inlet temperature from 2000° to 2500° R reduced the missile-plus-booster weight about 35 percent.

The effect of reducing both compressor and turbine efficiency is shown in figure 27. The lower curve is for the efficiencies which were used for all other calculations, 0.85 for both the compressor and turbine. In the upper curve, the compressor efficiency was reduced to 0.75 and the turbine efficiency to 0.80. By sacrifices in component efficiency, turbojet engines could be built more cheaply, but figure 27 demonstrates that use of turbojet engines with reduced efficiency considerably increases the total weight of missile plus booster, even for the shorter ranges. Such decreases in efficiency would therefore increase the over-all cost of the missile far more than any reduction to be gained in engine cost.

As with the ram-jet engine, the turbojet exhaust nozzle must have good efficiency as shown by figure 28. A decrease in velocity coefficient from 0.975 to 0.95 increases the weight 20 to 30 percent. It becomes of prime importance, therefore, to have an exhaust nozzle that is capable of efficient operation during boost over a range of Mach numbers and also at peak efficiency during cruise operation. Similar demands are also made on the engine inlet. It will be recalled that the turbojet engine was assumed to have continuously variable inlet and outlet geometry. Methods of achieving efficient operation of both inlets and outlets for turbojet engines over a range of conditions will be discussed in another paper.

From this brief survey of turbojet-engine requirements for supersonic propulsion, it can be seen that the most important factors are

light weight, good component efficiency, and if possible, higher turbine-inlet temperatures.

Rocket

Brief consideration will be given to a study of rocket-engine requirements for long-range missiles. It is well known that rocket missiles must be staged in order to avoid excessive weights at long ranges. It is also desirable to have shorter-range rockets available. A single rocket unit was therefore analyzed. This unit could be used as a single-stage rocket for short distances, and in combination with similar units would provide a good long-range two-stage rocket missile. A ballistic missile of this type is illustrated in figure 29. The staged missile consists of three units, each having similarly designed motors and fuel tanks. The only difference in the motors is the expansion ratio of the exhaust nozzles. The centerbody also contains the guidance equipment and 5000 pounds of pay load. All motors are started initially, but fuel is used only from the two outside or auxiliary units. When the fuel tanks in these units are empty, the units along with their motors are discarded, leaving the centerbody with a full fuel load. After the fuel is exhausted from the centerbody, the nose cone separates and continues on the path to the target. Being a complete unit in itself, the centerbody with minor modifications can be used as a single-stage short-range rocket missile. The nose cone of the centerbody contains the warhead along with coolant and ballast which are necessary for reentry of the cone into the atmosphere at the end of the flight. The shape of the centerbody nose cone shown in figure 29 does not represent the true slender shape which would be required in practice. The parallel arrangement illustrated has two advantages over the more conventional series arrangement: maximum thrust is obtained initially, and all motors can be started before launching to insure satisfactory motor operation.

The flight path for the two-stage rocket is shown in figure 30. Because of the uncertainties associated with the aerodynamics of high-speed glide rockets, only the ballistic path was considered in this paper. There are of course some uncertainties regarding the performance of ballistics rockets, such as the problems associated with reentry into the atmosphere. In the present analysis an attempt was made to allow for proper cooling load and ballast that were added to insure reasonable cone temperatures, cone stability, and moderate deceleration of the warhead during reentry of the cone into the earth's atmosphere. The missile is launched vertically, then turns after passing through most of the atmosphere. The booster units are discarded at an altitude of about 35 miles and burnout of the second stage occurs at an altitude of about 200 miles. Maximum height of the trajectory is 300 miles and the velocity during most of the path is over 20,000 feet per second.

2546-G

The performance of the ballistic rocket is presented in figure 31, which shows missile gross weight as a function of range for single and two-stage missiles using two different propellant combinations. Although the gasoline-oxygen two-stage rocket can be used for long ranges, the gross weights are very high. The most important factor affecting rocket performance is the specific impulse and although some improvement can be obtained by increasing the chamber pressure and improving the combustion characteristics, large improvements can be brought about best by using a propellant having a higher specific impulse.

Changing from gasoline-oxygen to ammonia-fluorine increases the specific impulse about 35 percent and reduces the gross weight to about one-half. One important feature to be noted with the ammonia-fluorine propellant is that for long ranges the weight does not increase rapidly with an increase in range and the two-stage missile with ammonia-fluorine propellant can be made to orbit about the earth with a gross weight of somewhat less than 300,000 pounds. The effect of the cooling and ballast loads on the performance of the ballistic rocket is appreciable because the cooling and ballast weight averaged 50 to 60 percent of the pay-load weight in most cases.

The effect of missile structural weight is presented in figure 32, which shows the effect of reducing the ratio of structural weight to gross weight by 10 percent. Although important, the improvements to be gained by reducing the structural weight are considerably smaller than those obtained by changing to propellants having a higher specific impulse. The structural weight is much less important because it is only a small part of the gross weight. Of course, if the structural weight were increased considerably because of less refined design, this weight could become very important.

In summarizing the performance of the ballistic rocket, it appears that the rocket missile is well suited to the extremely long ranges and that primary emphasis should be placed on the development of high-specific-impulse propellants.

Comparison of Engine Types

Although three different engine types have been considered, it is not the purpose of this analysis to compare these engines in an attempt to establish a superior power plant. It should be emphasized that the missiles utilizing these engines operate under different conditions, and, as was pointed out, small changes in the engine performance can cause large changes in the size and weight of the missile. However, a comparison of the weights of these missiles with the different engine types as already presented is interesting and informative provided caution is exercised in interpreting the results.

This comparison is presented on a gross weight and structural weight basis in figures 33 and 34. The structural weight as used in figure 34 is defined as gross weight of missile plus booster minus fuel and payload weight. The ram-jet and turbojet performance presented in these figures is based on the single-stage configurations discussed previously. The rocket performance is based on a two-stage ballistic trajectory with ammonia-fluorine propellant. Although every attempt was made to employ consistent assumptions wherever they could be applied to the different engine types and missile designs, the variations in the selected engine performance, types of missile design, and differences in the state of development could completely alter the comparisons. Each engine type exhibits different tactical, logistic, and technical characteristics which are beyond the scope of this discussion. In view of these arguments, the most important conclusion to be drawn from figures 33 and 34 is that all three engine types are competitive over the ranges investigated.

CONCLUDING REMARKS

The analysis has demonstrated that for supersonic missile propulsion, all of the engine variables become more important as the range increases. Although important weight savings can be realized by staging the ram-jet and turbojet missiles, single-stage designs can be considered practical for long ranges.

For the ram-jet engine, primary emphasis should be placed on increasing the efficiency of the components. Of utmost importance is the attainment of high combustion efficiency, whereas engine weight is of secondary importance. The most important turbojet-engine factors are light weight, good component efficiency, and high turbine-inlet temperature. Largest gains can be made in the rocket missile by utilizing propellants having high specific impulse.

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Figure 1

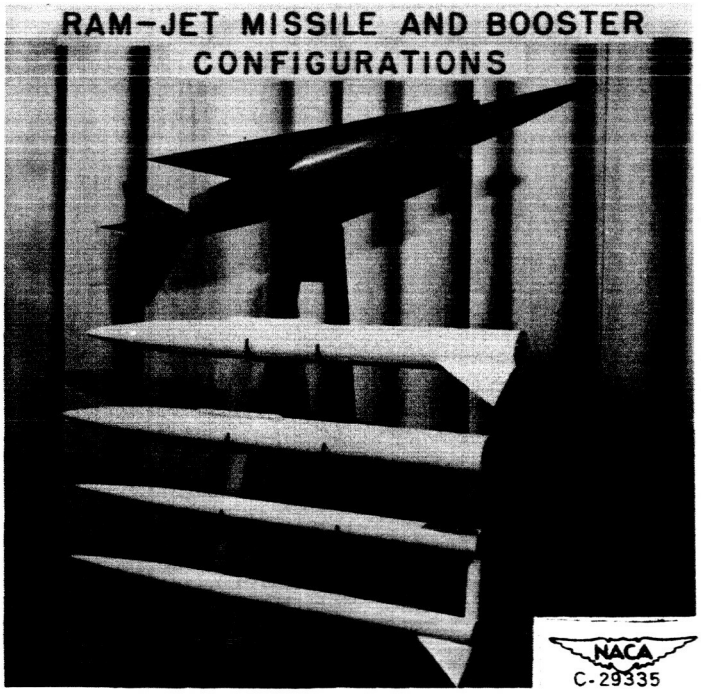

Figure 2

RAM JET MISSILE
FLIGHT PATH FOR ROCKET BOOST
TO FINAL MACH NUMBER

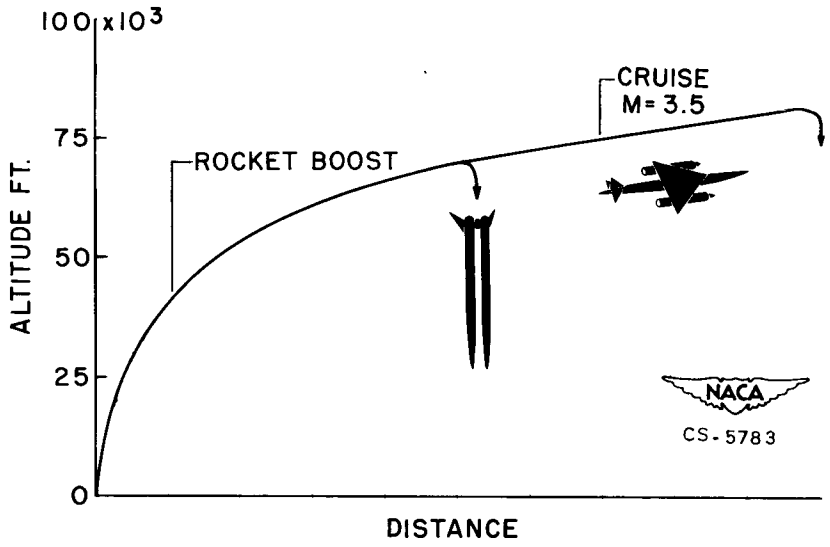

Figure 3

RAM JET MISSILE
EFFECT OF RANGE ON MISSILE
PLUS BOOSTER WEIGHT

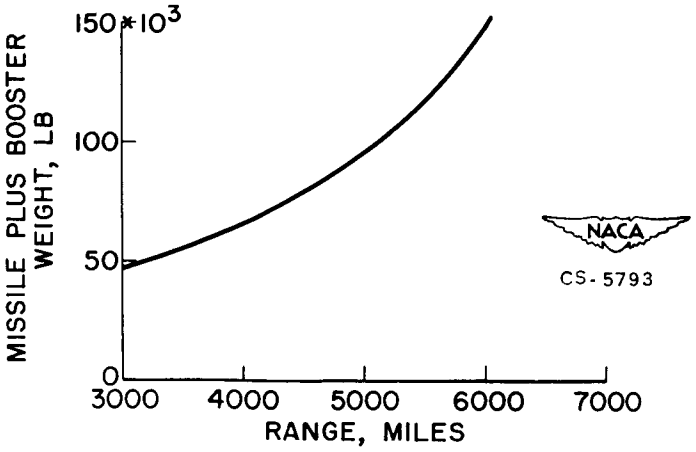

Figure 4

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RAM JET MISSILE
EFFECT OF MACH NUMBER AND ALTITUDE
RANGE, 6000 MILES

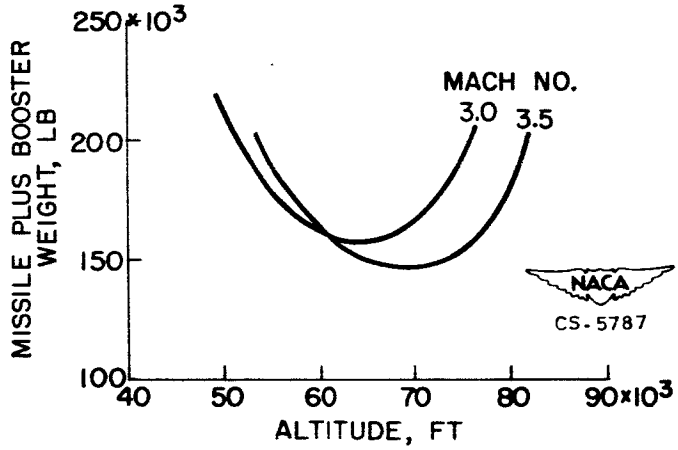

Figure 5

RAM JET MISSILE
EFFECT OF MACH NUMBER
RANGE, 6000 MILES

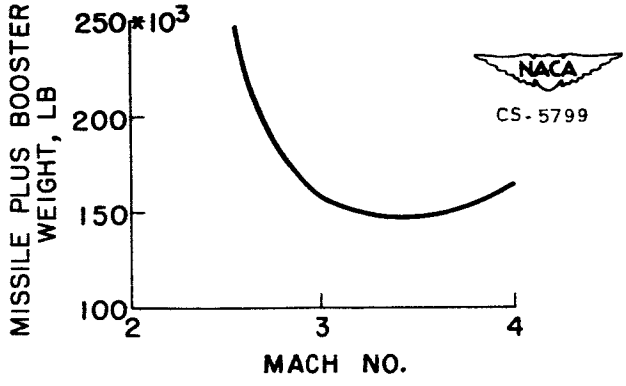

Figure 6

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RAM JET MISSILE
FLIGHT PATH FOR ROCKET BOOST
TO LESS THAN FINAL MACH NUMBER

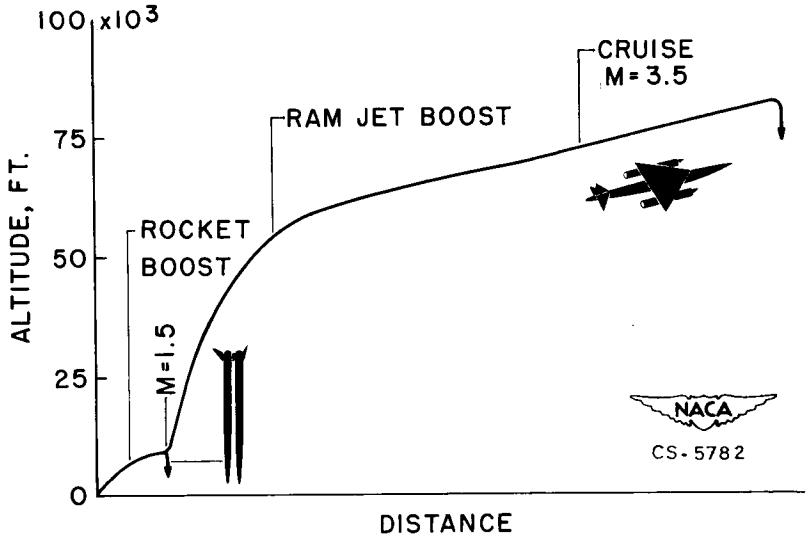

Figure 7

EFFECT OF RAM JET TAKE-OVER MACH NUMBER

RANGE, 6000 MILES CRUISE MACH NO., 3.5
ROCKET SPECIFIC IMPULSE, 240 SEC

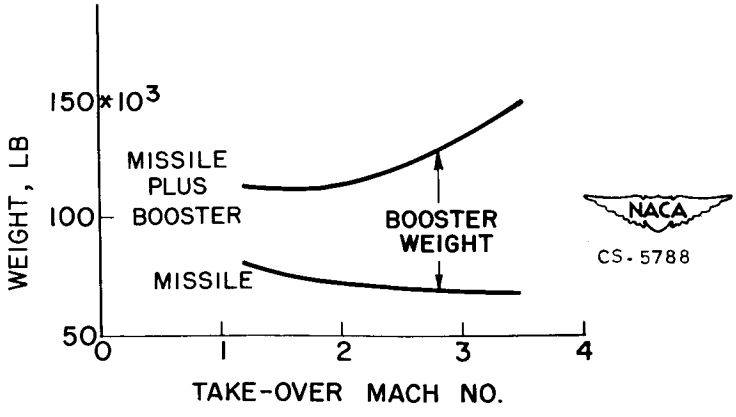

Figure 8

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RAM JET MISSILE
EFFECT OF ROCKET SPECIFIC IMPULSE

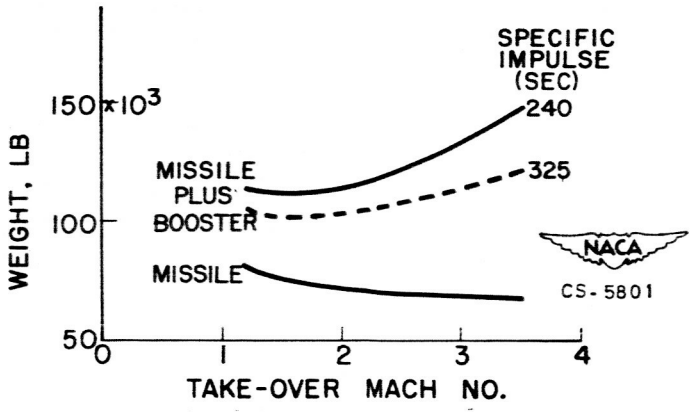

Figure 9

Figure 10

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TWO STAGE RAM JET MISSILE FLIGHT PATH
FOR ROCKET BOOST TO FINAL MACH NUMBER

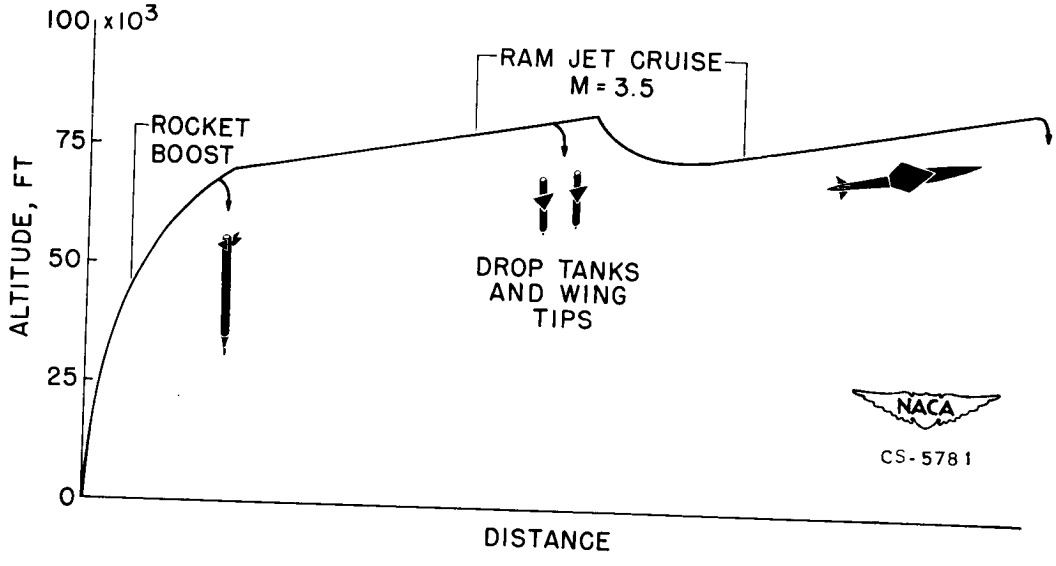

Figure 11

RAM JET MISSILE

EFFECT OF DROPPING WING SEGMENTS AND TANKS

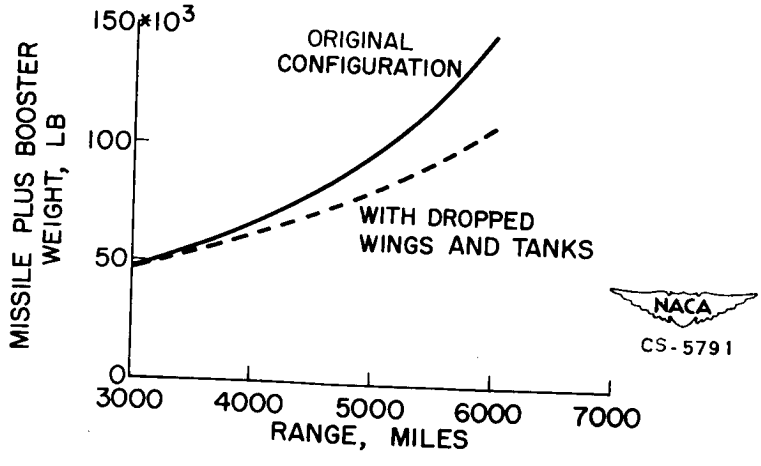

Figure 12

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RAM JET MISSILE
EFFECT OF COMBUSTION EFFICIENCY

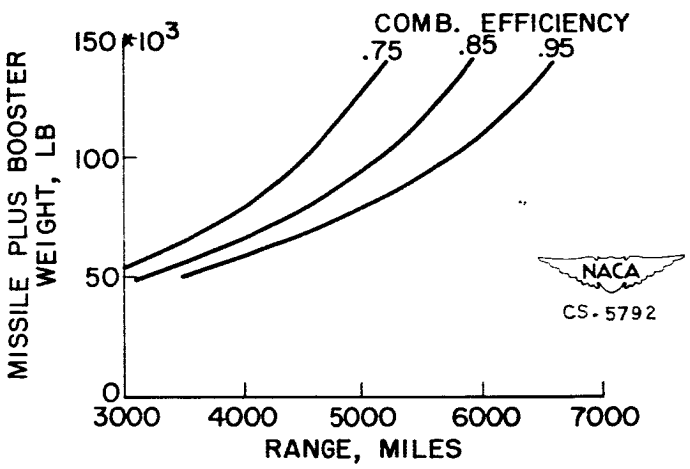

Figure 13

RAM JET MISSILE
EFFECT OF FUEL SELECTION

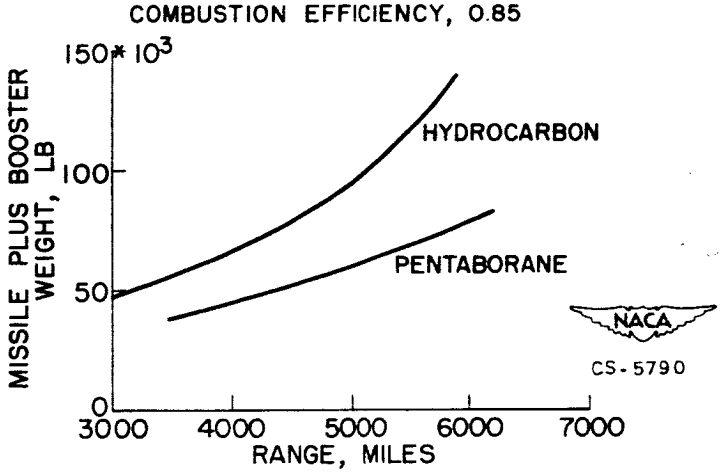

Figure 14

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RAM JET MISSILE
EFFECT OF ENGINE WEIGHT

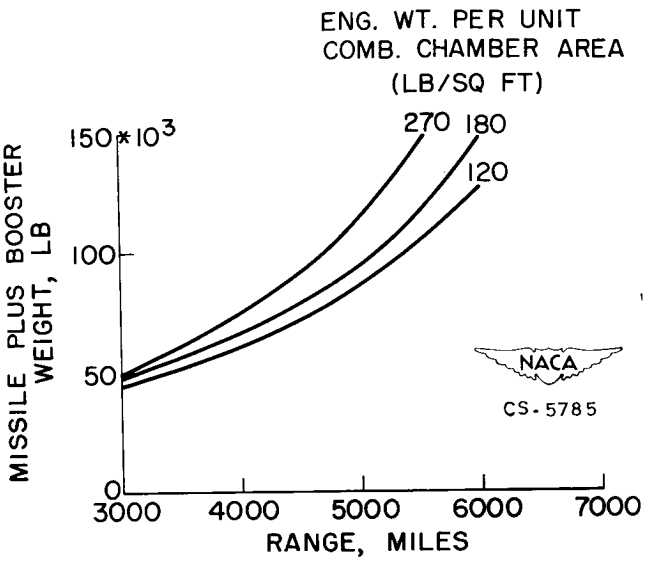

Figure 15

RAM JET MISSILE

EFFECT OF NOZZLE VELOCITY COEFFICIENT

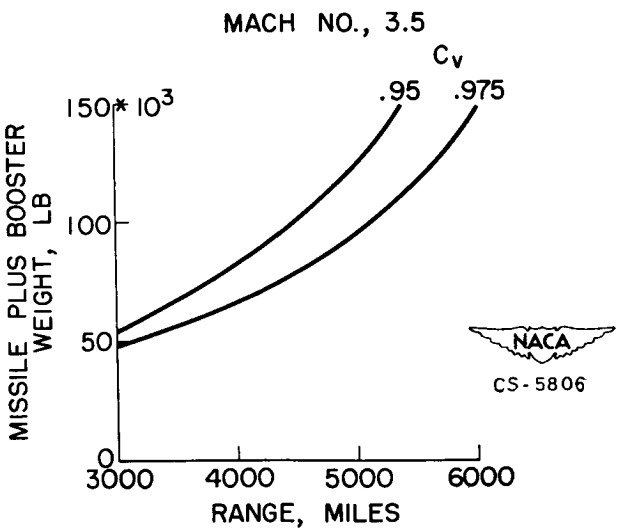

Figure 16

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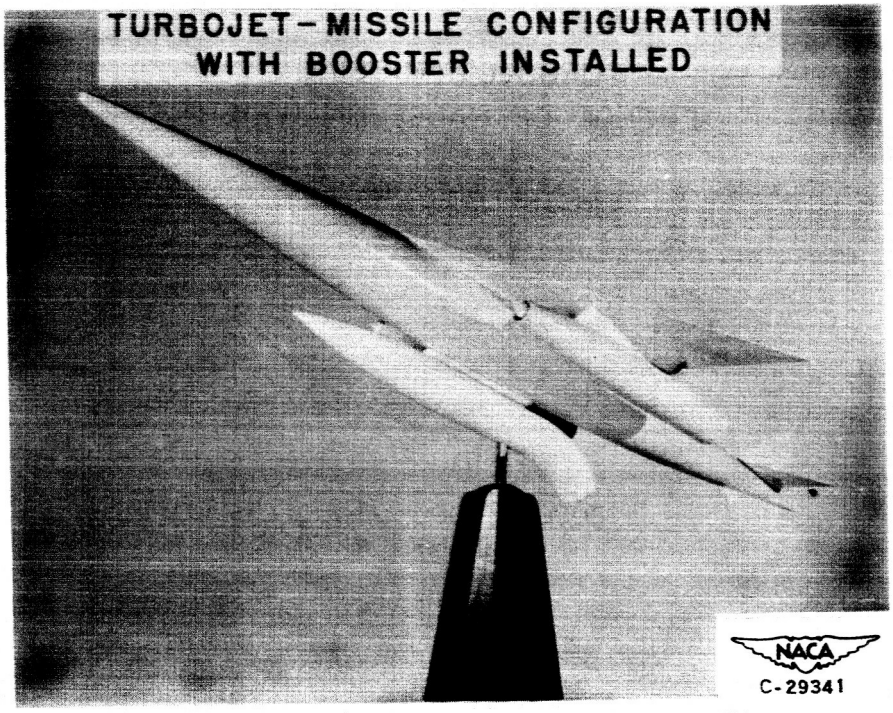

Figure 17

TURBOJET MISSILE
FLIGHT SCHEDULE DURING
CLIMB AND ACCELERATION

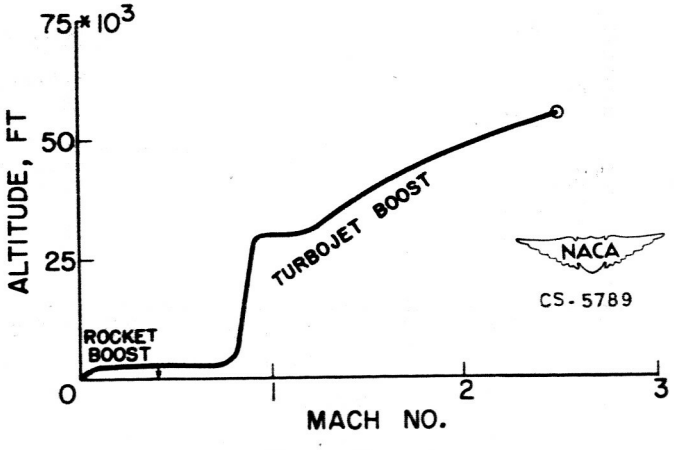

Figure 18

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TURBOJET MISSILE
EFFECT OF RANGE ON WEIGHT

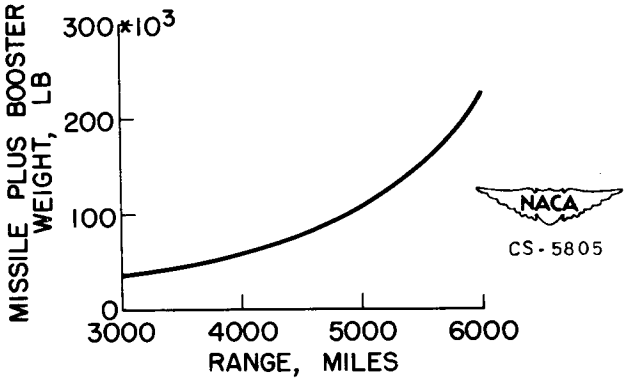

Figure 19

TURBOJET MISSILE
EFFECT OF MACH NUMBER

RANGE, 3000 MILES

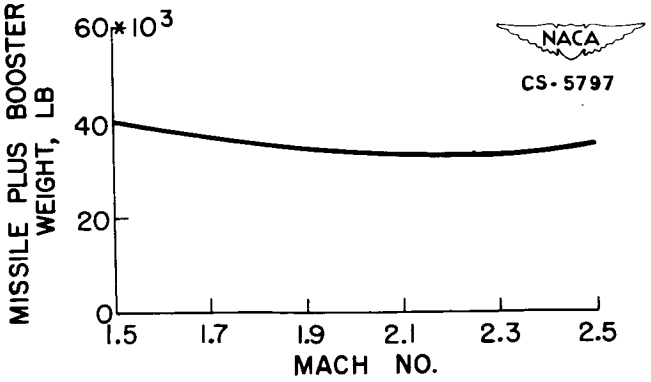

Figure 20

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Figure 21

STAGED TURBOJET MISSILE
FLIGHT SCHEDULE

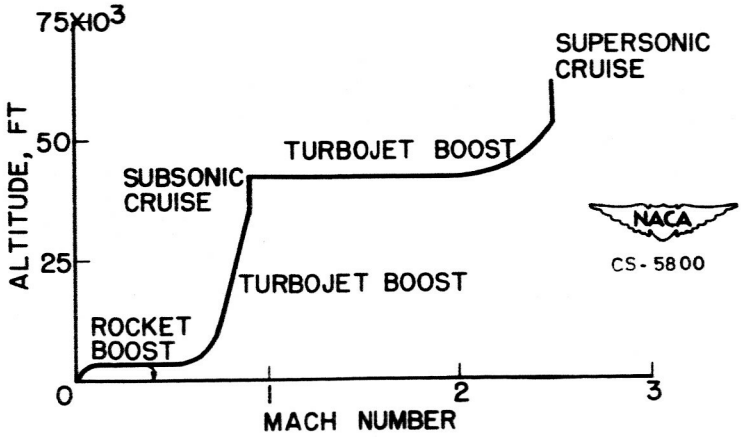

Figure 22

C

[redacted]

TURBOJET MISSILE
EFFECT OF STAGING

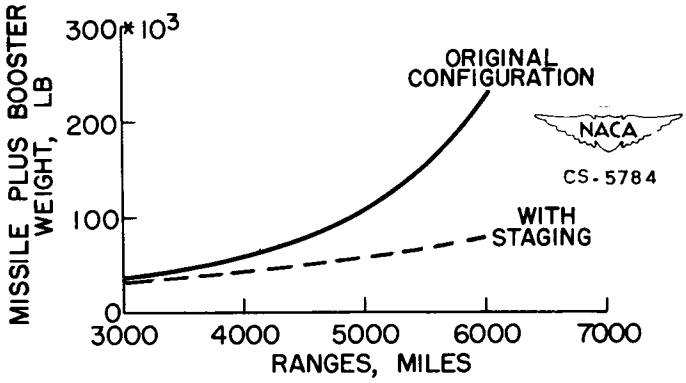

Figure 23

TURBOJET MISSILE
EFFECT OF COMPRESSOR PRESSURE RATIO

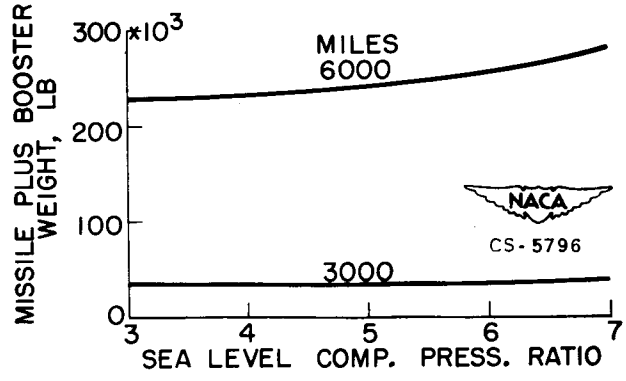

Figure 24

[redacted]

DECLASSIFIED

TURBOJET MISSILE
EFFECT OF ENGINE WEIGHT

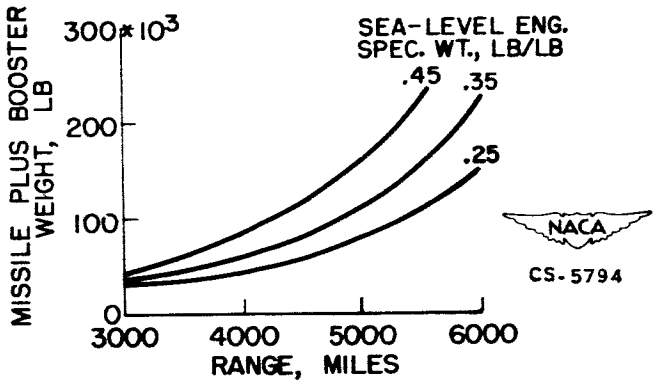

Figure 25

TURBOJET MISSILE
EFFECT OF TURBINE INLET TEMPERATURE

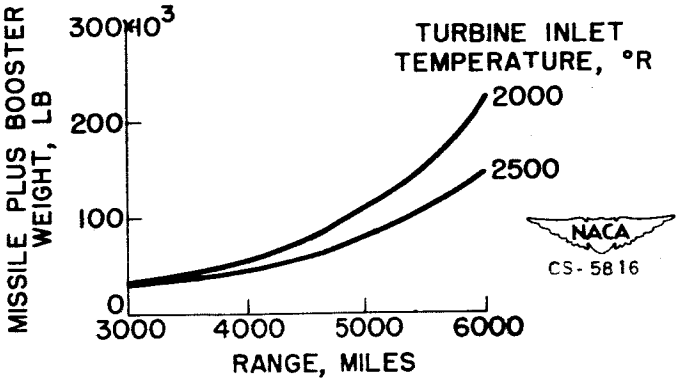

Figure 26

[redacted]

TURBOJET MISSILE
EFFECT OF COMPRESSOR AND TURBINE EFFICIENCY

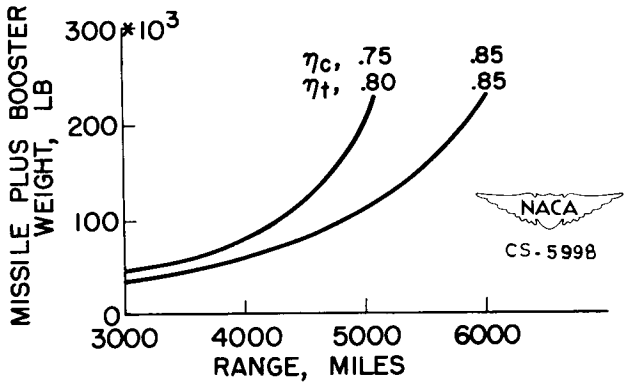

Figure 27

TURBOJET MISSILE
EFFECT OF NOZZLE VELOCITY COEFFICIENT

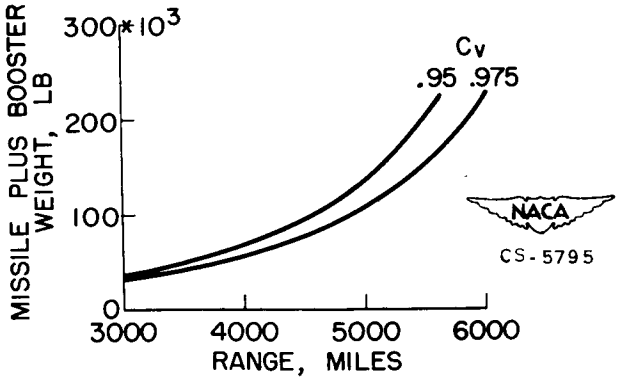

Figure 28

[redacted]

DECLASSIFIED

Figure 29

ROCKET BALLISTIC MISSILE
FLIGHT PATH FOR TWO STAGE MISSILE

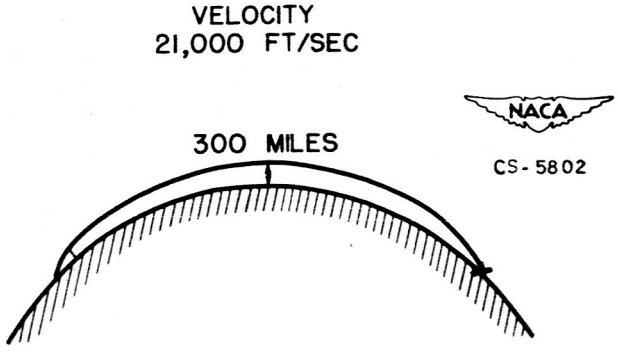

Figure 30

[redacted]

ROCKET BALLISTIC MISSILE
EFFECT OF PROPELLANT

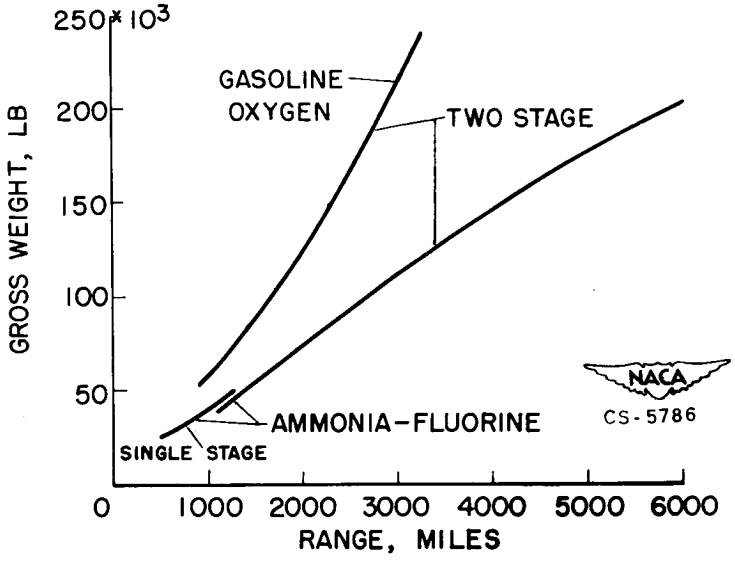

Figure 31

ROCKET BALLISTIC MISSILE
EFFECT OF STRUCTURAL WEIGHT

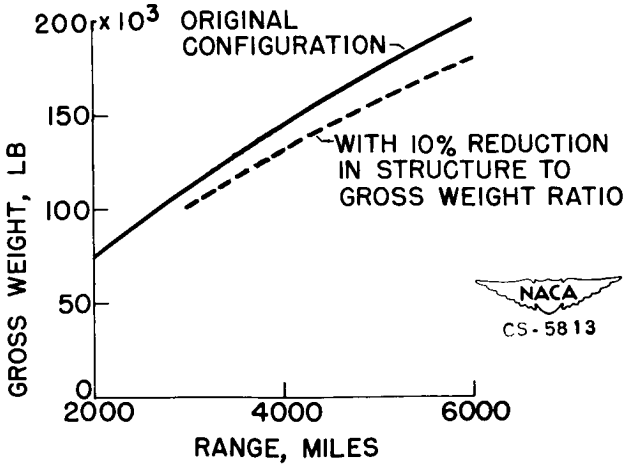

Figure 32

[redacted]AL

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2546-B

EFFECT OF ENGINE TYPE ON GROSS WEIGHT

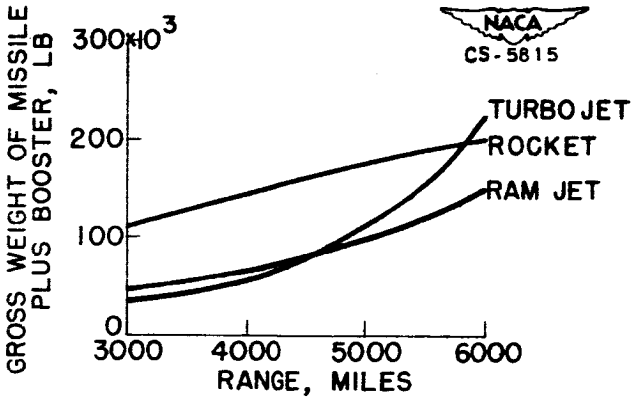

Figure 33

EFFECT OF ENGINE TYPE ON STRUCTURAL WEIGHT

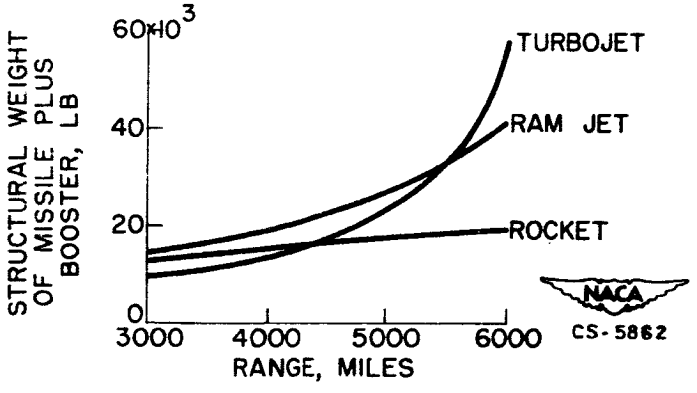

Figure 34

UNCLASSIFIED

3. - COMMENTS Ib

By E. J. Manganiello

In the analysis just presented the ram-jet engine was shown to possess considerable potential for the propulsion of long-range supersonic missiles, and emphasis was directed toward the importance of high combustion efficiency, the achievement of which is rather difficult under the adverse combustion chamber conditions existing at the high flight altitudes of interest in missile propulsion. During the past seven years an appreciable amount of effort has been expended by organizations throughout the country on research and development aimed at obtaining good combustion efficiency and over-all performance of the ram-jet engine. The experimental techniques and methods utilized in this work are interesting in themselves and it may be well to consider them briefly before proceeding with a discussion of the research results.

2546-C

The NACA has used four general experimental techniques in its ram-jet research. The first of these is the so-called "connected-pipe" technique wherein a ram-jet engine or only its combustor is directly connected to air compressor and exhauster equipment which provides temperatures, pressures, and velocities at the inlet of the combustor corresponding to those encountered under flight conditions. This method is useful for studying the combustion problems of the ram-jet engine but obviously cannot provide information pertinent to the diffuser performance or to the interaction effects between the diffuser and the combustor.

The second method, the "free-jet" technique, permits investigation of both the diffuser and combustor performance and their interaction. This method, which has been used by both the Langley and Lewis laboratories, consists of installing the ram-jet engine with its inlet immersed in a jet of air issuing from a supersonic nozzle. The supersonic jet must be large enough to supply both the internal air requirements of the engine and about an equal spillage around the cowl of the engine inlet. The free-jet technique provides information on the internal performance of the engine; however, it does not furnish information on the external aerodynamic characteristics of the engine. These can best be obtained in supersonic tunnels or in flight.

In supersonic tunnels such as the 8- by 6-foot tunnel, complete ram-jet engines have been tested over ranges of Mach number and angle of attack and both internal performance and external aerodynamic characteristics data were obtained under closely controlled conditions.

[redacted]

[redacted]

Several variations of the flight test technique have been used for ram-jet research. One method consists simply of mounting a ram jet on some suitable portion of an airplane, such as under the wings or fuselage, and studying its performance under different flight conditions. The flight conditions are, of course, limited by the capabilities of the airplane. Another variation which has been utilized with successful results is the "drop test" technique wherein a ram-jet engine model is carried to altitude by a carrier aircraft, started, and then released from the airplane. The ram-jet missile accelerates under the combined effects of gravity and ram-jet thrust and provides data over a range of altitude and Mach number.

2546-C

A third flight method is the ground launched rocket-boost technique. In this method, a ram-jet powered missile model is launched and boosted to a suitable flight speed whereupon booster separation occurs and the ram-jet model continues under its own power on a prescribed flight plan. This method has been used extensively by the Langley laboratory at its Wallops Island installation, and in one such test the ram-jet missile attained an altitude of 67,000 feet under ram-jet power. At this time the fuel was exhausted; however, the model coasted up to an altitude of 159,000 feet. The maximum Mach number attained in this flight was 3.12.

In the above flight methods, data on diffuser pressure recovery, ram-jet thrust, missile drag, and other characteristics are obtained by means of telemetering and radar tracking. This technique is very useful for obtaining data in the transonic region.

In the next paper, a discussion of some of the results of such ram-jet work will be presented.

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DECLASSIFIED

4. SOME RAMJET ENGINE PROBLEMS

By Roland Breitwieser, Eugene Perchonok

and Thomas B. Shillito

C[redacted]

DECLASSIFIED

4. - SOME RAM-JET ENGINE PROBLEMS

By Roland Breitwieser, Eugene Perchonok, and
Thomas B. Shillito

Analysis of the ram-jet-powered long-range missile has shown the range of conditions over which the ram jet must operate and what component efficiencies are required. The purpose of this discussion is to show if and by what means the required ram-jet performance can be achieved.

2546-D

Combustion efficiency is probably the most important ram-jet performance criterion. The previous analysis indicated that combustion efficiencies of 85 percent or higher are required if excessive missile weights are to be avoided. It is therefore necessary to examine in detail the operating conditions of the assumed long-range vehicle to see if the required combustion efficiencies can be achieved with existing combustors. The geometrical layout and operating conditions of two fixed-geometry ram-jet engines optimized according to the preceding analysis for Mach numbers of 3 and 3.5 are shown in figure 1. (In this figure there are two reference conditions: station 0, the free-stream condition, and station 2, the burner-entrance condition.) The Mach number 3 ram-jet engine initiates its cruise flight at an altitude of 62,500 feet and terminates its flight at an altitude of 87,500 feet. Because of its higher flight speed, the Mach number 3.5 ram-jet engine initiates its cruise at a higher altitude, 70,000 feet, and terminates its flight at 94,000 feet.

It is interesting to note the similarity of the combustor-inlet conditions. The combustor-inlet pressure for the Mach number 3 ram jet varies from about 20 to 6 pounds per square inch and for the Mach number 3.5 ram jet, varies from 21 to about 7 pounds per square inch. The velocities are the same, 270 feet per second. The inlet temperature for the Mach number 3 ram jet is 1100° R; the Mach number 3.5 ram jet has a higher stagnation temperature, 1350° R. At both flight conditions, the fuel-air ratio operating range is of the order of 0.02 to 0.03. Data are not available for a single combustor evaluated over these conditions; several specific combustors have been evaluated at some of the more severe conditions along the assumed flight paths.

For example, a piloted V-gutter combustor has been evaluated at the following combustor-inlet conditions (reference 1): static pressure, 16 pounds per square inch; temperature, 1060° R; and velocity, 240 to 260 feet per second (fig. 2). These conditions correspond roughly to the midpoint of the Breguet cruise path for the Mach number 3 ram jet. Combustion efficiencies of 85 percent or higher were achieved at fuel-air ratios as lean as 0.015.

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The type of configuration used to develop this high combustion efficiency at low fuel-air ratios is illustrated in figure 2. The piloted V-gutter combustor and the V-type flame holders are located at the aft end of the centerbody. The air for the piloted burner is taken through an inlet scoop ahead of the fuel sprays. The fuel spray system is conventional; however, a mixture control sleeve shrouds the fuel-air mixing region downstream of the injector. The purpose of the fuel control sleeve is to stratify the air so that nearly ideal mixtures are delivered to the flame-holding elements.

Another combustor has also been evaluated over some of the conditions indicated in figure 1 (reference 2). In this case (fig. 3), the test conditions correspond to a portion of the assumed Breguet cruise path. With this configuration, a can-type combustor as illustrated, combustion efficiencies as high as 85 percent could be maintained to the low pressures corresponding to the end of the assumed flight path. This specific combustor was designed for rich operation and peak efficiency at fuel-air ratios from 0.05 to 0.06. Since near-stoichiometric mixtures are desirable for efficient burning, achievement of high efficiency at leaner over-all mixtures requires that the fuel-air charge be stratified and that air be bypassed around the flame-holder region. An example of how this has been achieved with a can-type combustor is shown in figure 4. These data are from the Applied Physics Laboratory of the Johns Hopkins University (reference 3). The combustion efficiency is presented for two configurations: one a typical 18-inch can-type combustor with a single injector, and the other a multiple fuel-injector system in a simulated can combustor. With the single injector system, the combustion efficiency peaked over a very narrow band of fuel-air ratios; when a multiple fuel injector system was used, however, it was possible to achieve an essentially constant combustion efficiency over a wide fuel-air ratio range. This was accomplished by stratifying the fuel-air mixture. One injection system was used for low fuel-air ratio operation; and a second injection system was employed to extend the fuel-air ratio range to richer over-all values.

The effect of controlled fuel distribution on the performance of a piloted V-gutter combustor is illustrated in figure 5 (reference 1). Two types of injection system were evaluated, outer-ring injection and inner-ring injection. A lean fuel-air ratio limit of 0.025 was observed for outer-ring injection. By injecting fuel through the inner ring to provide a more combustible mixture in the region of the central pilot burner, the lean limit was extended to 0.015. If mixture controlling sleeves or shrouds are employed, the lean limits may be lowered to 0.02 and 0.01.

Thus a design technique by means of which the performance of both can and V-gutter flame holders can be optimized at a desired fuel-air ratio has been demonstrated.

2546-D

Data for several combustors at some of the combustor-inlet conditions that will be encountered with the assumed missile are available, but data for a single combustor over the entire flight path are not available. Although the fuel distribution system can be designed to give high combustion efficiency at desired fuel-air ratios, the effects on the combustion efficiency must be determined for some of the other inlet parameters, namely, inlet velocity, temperature, and pressure. These effects must be known to estimate performance of the combustors at conditions other than those checked by experiment.

2546-D

The effect of velocity on combustion efficiency is shown in figure 6. The temperature, pressure, and fuel-air ratio were constant for all of the combustors examined but not necessarily at the same level, so these data should be used not to indicate the performance level, but rather to determine the trend of combustion efficiency with respect to inlet velocity. A regular, consistent trend of a decrease in combustion efficiency with an increase in combustor-inlet velocity is observed for both 5-inch (unpublished NACA data) and 28-inch V-gutter combustors (reference 4).

The consistent variation of combustion efficiency with inlet conditions for the variety of combustors is further indicated in figure 7. Here the effect of combustor inlet pressure on combustion efficiency is indicated; the fuel-air ratio, temperature, and velocity are constant for each configuration. The data included are for a 20-inch can combustor (reference 2), a 20-inch V-gutter combustor (reference 5), a 5-inch V-gutter combustor (unpublished NACA data), and a 5-inch oxygen-hydrogen piloted combustor (unpublished NACA data). The effect of increasing pressure is to improve the combustion efficiency of all combustors investigated. Not only is the trend established, but because of the consistency with a wide variety of combustors, the general order of magnitude can be established.

The effect of inlet-air temperature on combustion efficiency is shown in figure 8. As the inlet-air temperature was raised, a regular increase in the combustion efficiency was observed for a series of combustors including a 5-inch (unpublished) and a 20-inch V-gutter (reference 5) and a 20-inch can combustor (reference 2). This increase in combustor-inlet temperature corresponds, of course, to an increase in flight Mach number.

Applying the well established trends of combustion efficiency variations with inlet conditions permits the estimation of the performance of combustors at conditions other than those checked by actual experiment.

Satisfactory performance has been indicated for a combustor at the highest altitude condition of the proposed missile flight. Reducing the altitude, that is, going to regions of higher pressure, should

improve the combustion efficiency. An increase in Mach number from 3 to 3.5 increases the inlet temperature and, again, should improve the combustion efficiency. The combustion efficiency of 85 percent assumed for the proposed missile therefore appears to be quite realistic and should be achievable with combustors not greatly different from existing combustors.

Thus far only the isolated ram-jet combustor has been considered. However, there are other problems that enter where the complete ram-jet engine is concerned. For example, the problems of flow instability, diffuser pusling, or changes in angle of attack must also be considered. It has been observed that when a ram-jet engine is taken to angle of attack, the combustor performance is affected (fig. 9). The data shown were obtained in the Lewis 8X6 supersonic tunnel with a 16-inch ram-jet engine (reference 6).

As the angle of attack is increased from 0° to 6° and then to 10°, the combustion efficiency progressively drops. At a given fuel-air ratio the combustor-inlet pressure, temperature, and velocity were essentially constant; hence, this effect is probably caused by the distorted combustor-inlet air-flow pattern. This distorted air-flow pattern was verified by a velocity survey which showed a shift in velocity profile at the entrance of the combustor as the engine went to angle of attack. This distortion of air-flow pattern undoubtedly causes a maldistribution of local fuel-air ratios entering the combustor and consequently reduces the combustion efficiency. Presumably, the combustor performance could be improved at angle of attack by developing a type of fuel injection system which would compensate for this maldistribution of air flow, but at best this would be a rather awkward technique.

It appears to be more logical to avoid this problem by developing inlet diffusers that are less sensitive to angle of attack operation. In the past, emphasis in diffuser design has been given to high pressure recovery and low drag features, with little attention given to the performance at angle of attack and to velocity distribution in the region of the combustor inlet (diffuser exit). It is obvious that these latter two features must also be considered in the design of supersonic diffusers for ram jets.

Another problem associated with the complete ram-jet engine is the problem of flow instability. Although the origin of the instability can occur within the engine, for example, in the combustor, it is generally conceded that the diffuser usually initiates ram-jet flow pulsations. The amplitude and frequency of the pulse initiated depends upon the complete engine geometry, not merely upon the type of diffuser employed. An example of the unstable flow observed with a 16-inch ram jet is presented in figure 10. The frequency of the pulse in cycles

per second and the maximum amplitude of the pulse, referred to the free-stream total pressure, are plotted as a function of the mass flow ratio. With the type of inlet employed, pulsing is initiated if the mass flow is reduced below the critical value. Moreover, the amplitude of the pulse appears to increase linearly as the mass flow is reduced, until at a mass flow of 50 percent of the maximum, amplitudes as large as 70 percent of the free-stream total pressure were observed.

There are several theories that attempt to predict the amplitude and frequency of these variations. A representative prediction based upon the Helmholtz resonator analogy and proposed in reference 7 and unpublished NACA work is shown on figure 10. Experimental data and the theoretical line correlate quite well for both frequency and amplitude.

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The discussion thus far has been concerned with cold-flow instability. In the burning case, the combustor is usually blown out if the diffuser is operated subcritically. One case has been observed (fig. 10) in which not only did the combustor continue to operate at subcritical flow conditions, but the amplitude of the cold-flow pulse instability was reduced to a negligible value. However, such combustor configurations are few and their actions not fully understood. It is therefore generally desirable, if possible, to avoid subcritical engine operation.

Thus far, combustor operation has been discussed and the direction for improving combustor operation shown. Possible design trends for the diffuser and diffuser-combustor combination that should give improved performance have been indicated.

Gains can also be made in ram-jet engine performance by the use of special fuels. The ram jet is unusually well suited for the use of special fuels because of the absence of moving parts in the exhaust stream. This permits the use of those fuels that may generate corrosive or abrasive products without damage to ram-jet engine structural elements. Fuels that offer performance advantages over conventional hydrocarbon fuels are those fuels that are more reactive than hydrocarbon fuels or those that have higher heating values than hydrocarbon fuels. The heating values considered are: heating value per pound, which is valuable for maintaining a low fuel consumption; heating value per unit volume, which is desirable to reduce missile size; or heating value per unit weight of air, a high value of which is desirable for the high thrust application, that is, a high temperature rise across the combustor. The fuels that give these high heating values are generally called the high-energy fuels. The high-energy fuels consist primarily of the light metals and their hydrides.

Representative theoretical performance characteristics of some of these fuels evaluated in an idealized ram-jet engine, at a flight Mach number of 3.5, are shown in figure 11. Performance characteristics are

shown for pentaborane, boron, magnesium, and a slurry of magnesium compared with JP-3 (references 8 to 10). Pentaborane is a boron hydride, liquid at normal room temperatures, a very desirable fuel, but not available in any realistic quantity. Boron, a metal, is generally supplied in powder form and would usually be used in this form. Magnesium is readily available as a powder and presumably would be burned in this form.

The 50 percent suspension of magnesium in JP-3 is a paintlike material embodying many of the handling characteristics of conventional liquid fuels. The magnesium - JP-3 fuel is customarily called a slurry fuel, and consists of very finely divided magnesium suspended in a gelled hydrocarbon.

The data of the high-energy fuels are compared with a reference hydrocarbon, JP-3, and are presented as thrust, in pounds per pound of air, as a function of fuel consumption, in pounds of fuel per pound. Comparison of the fuels first at the low thrust level, 70 pounds per pound air per second, a condition corresponding to the thrust requirements of a long-range missile, is as follows: The pentaborane has a lower fuel consumption than JP-3. A previous paper on long-range missile analysis showed the lower fuel consumption of pentaborane applied to a long-range missile permits a 50-percent missile weight. The theoretical performance of boron also indicates weight and size savings for a long-range missile, although not so great as in the case of pentaborane. Magnesium fuels exhibit a considerably higher fuel consumption than the boron fuels at the low thrust levels, hence should not be used for this type of application. Magnesium fuels permit the attainment of thrusts higher than those available for a hydrocarbon fuel. The JP-3 fuel peaks at a thrust of about 160, and magnesium slurries or magnesium permit substantial thrust gains. Thus fuels are available that permit increased thrust or savings in fuel consumption. Because of the encouraging theoretical results, a program evaluating the combustion characteristics of the high-energy fuels is in process at this laboratory. A boron hydride, diborane, has been evaluated in a small-scale combustor and found to have combustion characteristics improved over those of the JP-3. Solid fuels, metals such as magnesium, aluminum, and boron, have been evaluated in the form of wires, powders, suspensions, and slurries of these metals in hydrocarbons. Boron exhibits about the same reactivity as a conventional hydrocarbon fuel. Magnesium not only gives higher thrust characteristics but also exhibits improved combustion stability because of its high chemical reactivity.

The indication of excellent combustion performance of the magnesium slurry in the small-scale equipment indicated the desirability of further combustion evaluation. Tests were therefore conducted with a slurry fuel, a 60 percent suspension of magnesium and JP-3 (reference 11). The fuel was evaluated at the following conditions: pressure, 15 pounds per square inch; temperature, 1660° R; and velocity, 340 to

450 feet per second. These conditions were chosen to simulate the entrance conditions of a jet-engine afterburner. These conditions can also be construed to simulate a ram jet at a flight Mach number of 4 and an altitude of 80,000 feet. The two types of injection system used in the 6-inch-diameter burner were wall injection and upstream injection, as shown in figure 12. The data are presented in terms of air specific impulse, as a function of fuel-air ratio against fraction of stoichiometric fuel-air ratio (equivalence ratio). The reference fuel, JP-3, was first evaluated with the wall injection system. The data presented in figure 12 indicate that the operation was very limited. The thrust level was of the order of about 145 seconds. The 60 percent suspension of magnesium evaluated in the same wall injection system permitted operation over the entire range of fuel flows. From the very low fuel flow rates up to the limits of the injection equipment, the slurry fuel burned stably. Corresponding with the increase in operational limits, there was a substantial thrust increase over the JP-3. The combustor configuration was altered to permit a more optimum environment for the JP-3 fuel by use of upstream fuel injection. Even with the optimized combustor, however, lower thrusts and a narrower band of operating limits were obtained for the JP-3 as compared with a magnesium slurry. Because of the encouraging results, a magnesium slurry was next applied to an actual ram-jet engine, shown in figure 13.

A 6.5-inch-diameter engine of the type used to power an aerodynamic test vehicle was selected (reference 12). The data presented in figure 13 are from a connected-pipe test setup for the slurry (unpublished data) and from free-jet tests for the hydrocarbon fuel (reference 12). The data are expressed as propulsive thrust coefficient as a function of actual fuel-air ratio divided by stoichiometric fuel-air ratio fraction, or equivalence ratio. Data for 50 percent magnesium slurry data indicate that nearly theoretical performance can be achieved. The coefficients of thrust for the magnesium slurry are about 17 percent higher than the theoretical values for hydrocarbon and 40 percent higher than actual hydrocarbon values. These data substantiate the thermodynamic performance predictions of this fuel.

It has been shown that with respect to combustion, the performance requirements for a ram-jet engine, for a long-range missile, should be achievable. Performance limitations are more apt to occur from problems such as angle of attack operation or flow instabilities. Some suggestions for future research to reduce the severity of these problems have been briefly indicated. The recent experimental confirmation of the thermodynamically indicated performance gains of high-energy fuels indicates that further research is merited in this field.

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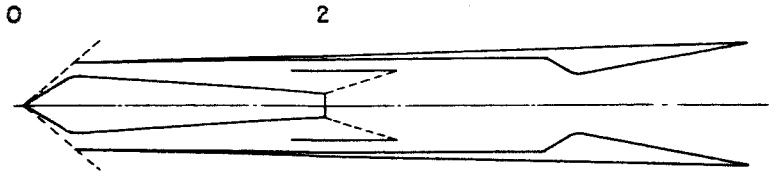
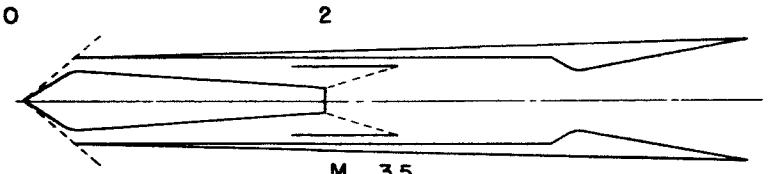
RAM-JET ENGINE OPERATING CONDITIONS



M_0, 3.0

ALTITUDE (FT)	P_2 (LB/SQ IN.)	T_2 (°R)	V_2 (FT/SEC)
62,500	19.8	1100	270
68,500	14.7	1100	270
87,500	6.0	1100	270



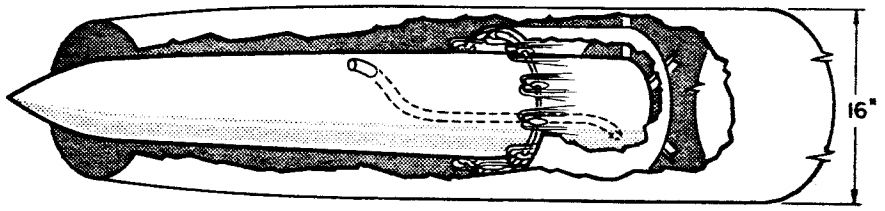
M_0, 3.5

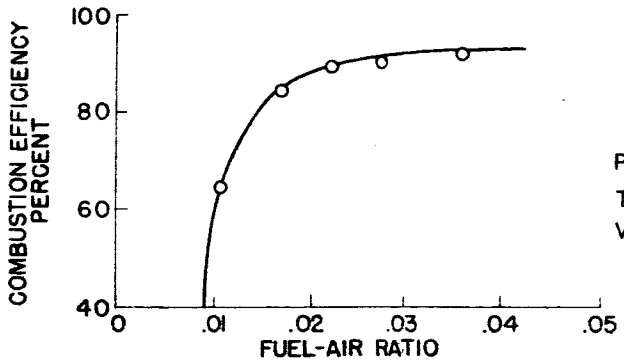
ALTITUDE (FT)	P_2 (LB/SQ IN.)	T_2 (°R)	V_2 (FT/SEC)
70,000	21.5	1350	270
78,000	14.7	1350	270
94,000	6.9	1350	270

NACA
CS-5807

Figure 1

PILOTED V-GUTTER COMBUSTOR





NACA
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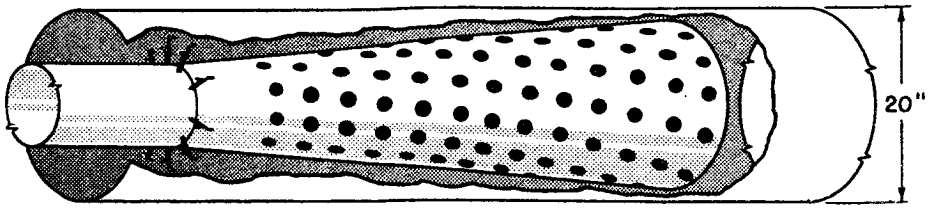
P_2 16 LB/SQ IN.
T_2 1060° R
V_2 240-260 FT/SEC

[redacted]

CAN TYPE COMBUSTOR (CONVAIR)



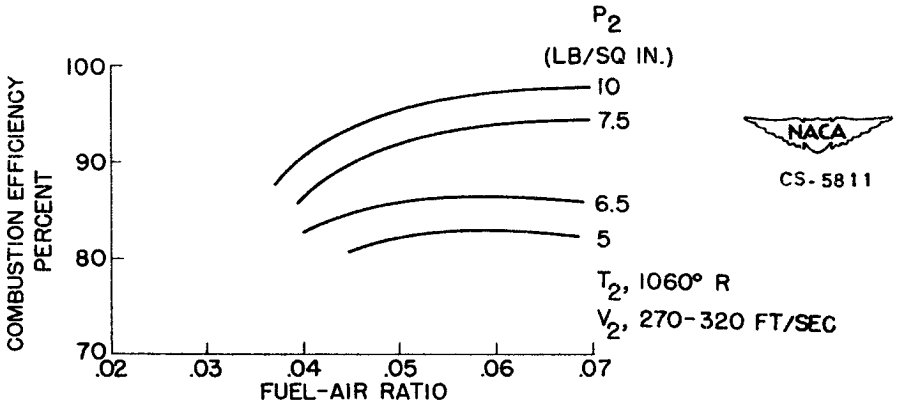


Figure 3

CONTROLLED FUEL DISTRIBUTION - CAN TYPE COMBUSTOR (APL/JHU)

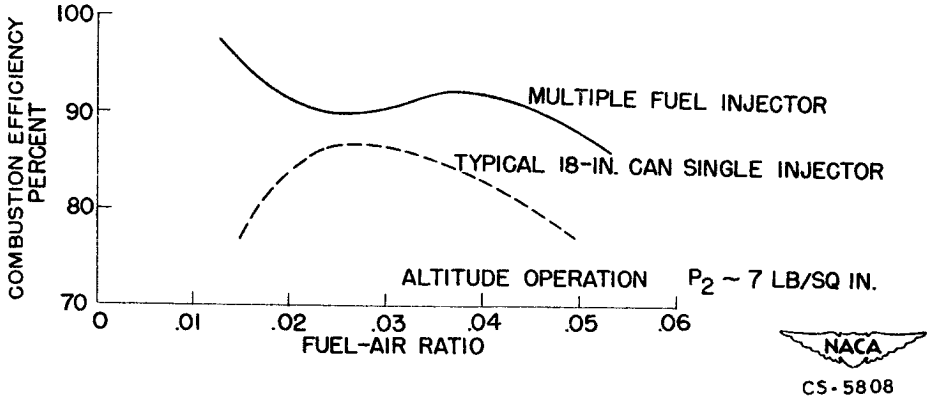


Figure 4

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CONTROLLED FUEL DISTRIBUTION - PILOTED V-GUTTER COMBUSTOR

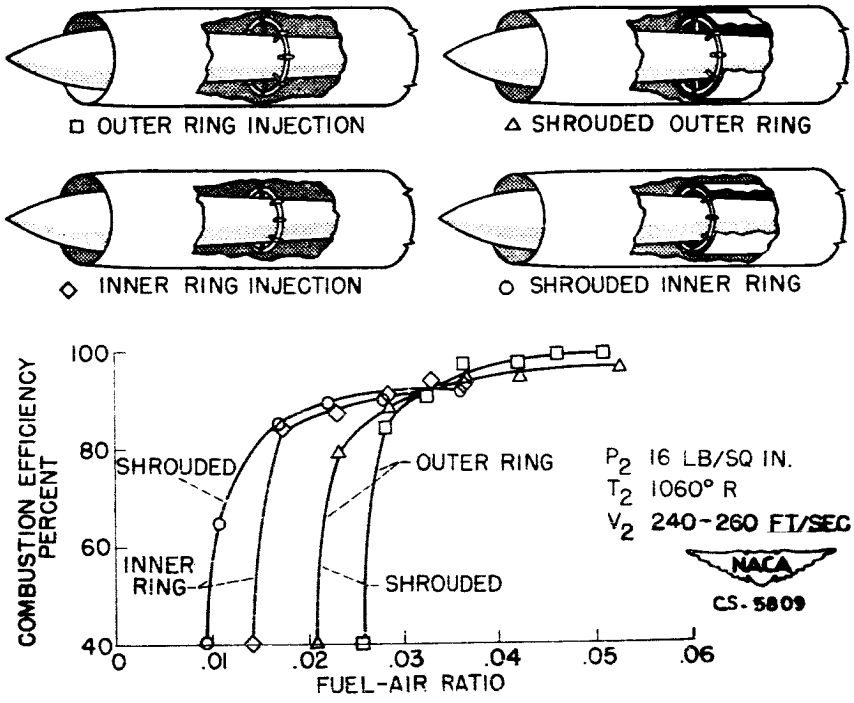

Figure 5

EFFECT OF VELOCITY ON COMBUSTION EFFICIENCY

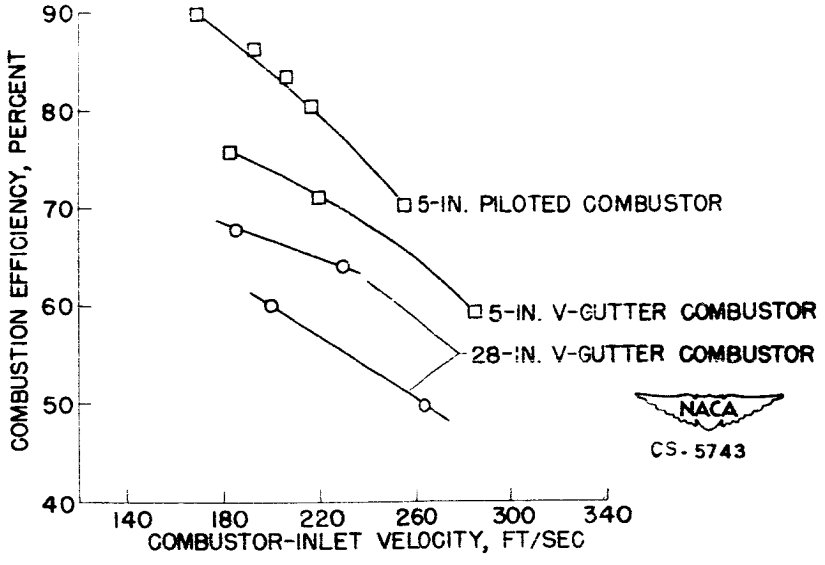

Figure 6

[redacted]

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EFFECT OF PRESSURE ON COMBUSTION EFFICIENCY

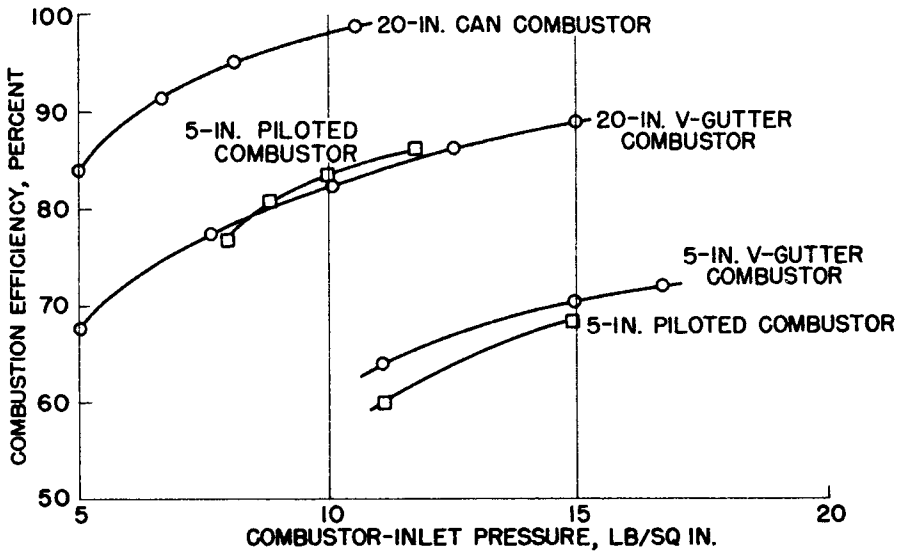

Figure 7

EFFECT OF TEMPERATURE ON COMBUSTION EFFICIENCY

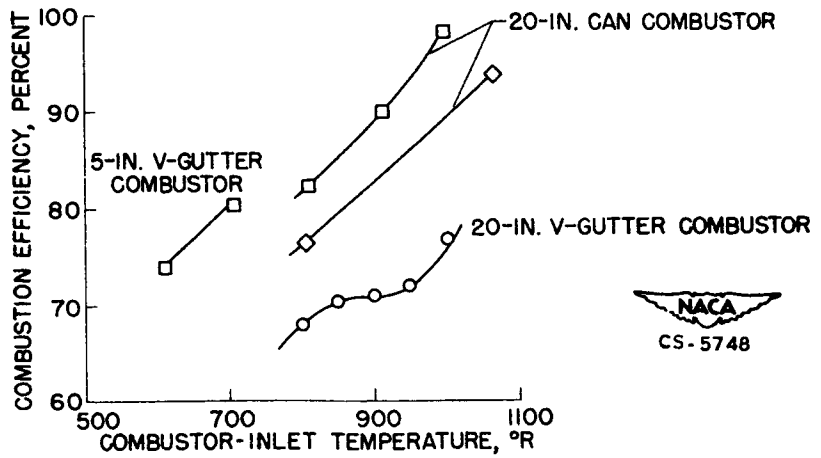

Figure 8

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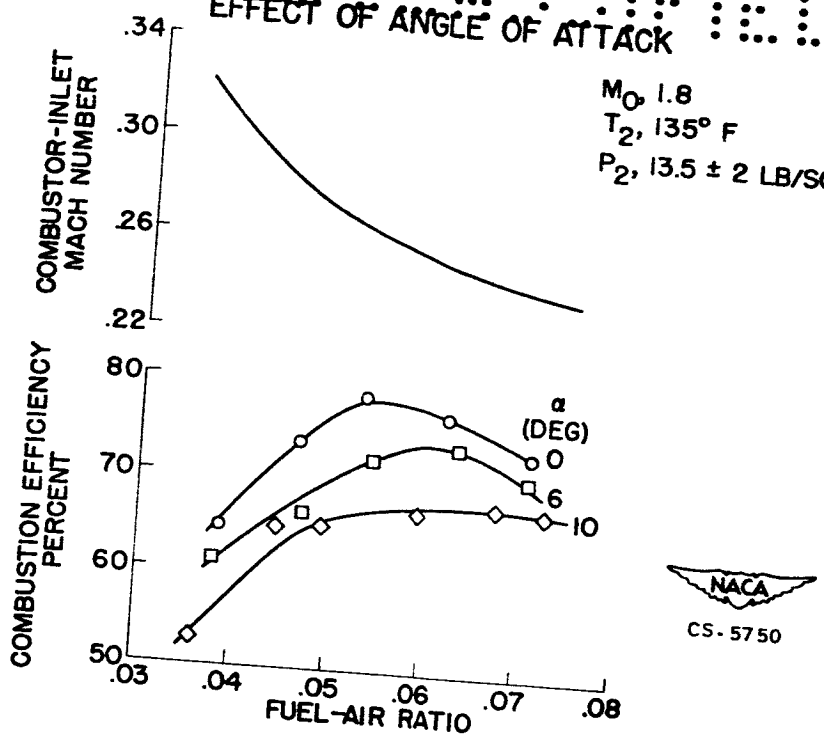

Figure 9

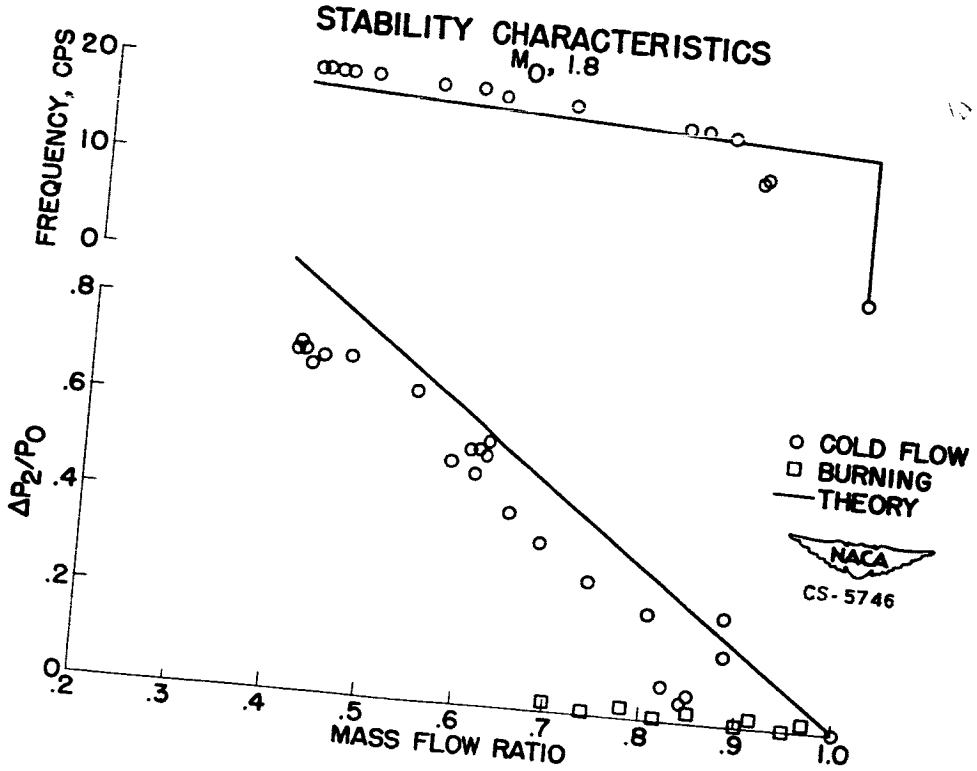

Figure 10

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THEORETICAL RAM-JET PERFORMANCE OF HIGH ENERGY FUELS

M_0, 3.5

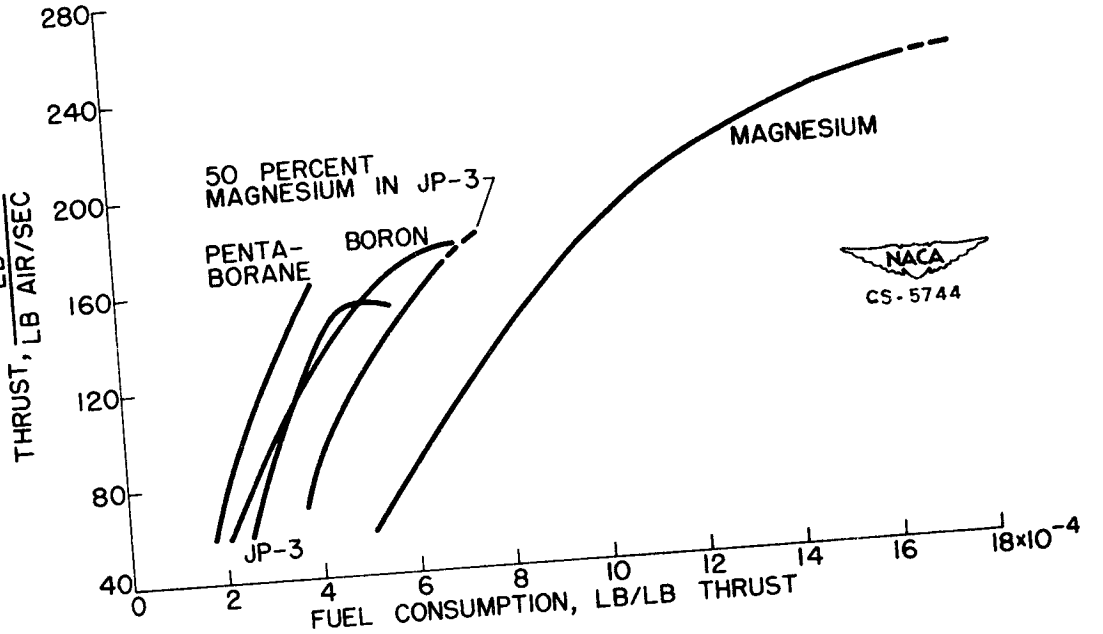
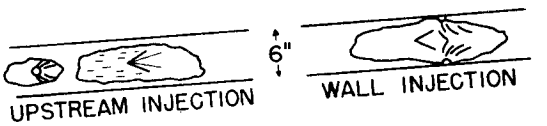


Figure 11

EFFECT OF MAGNESIUM ON COMBUSTOR PERFORMANCE



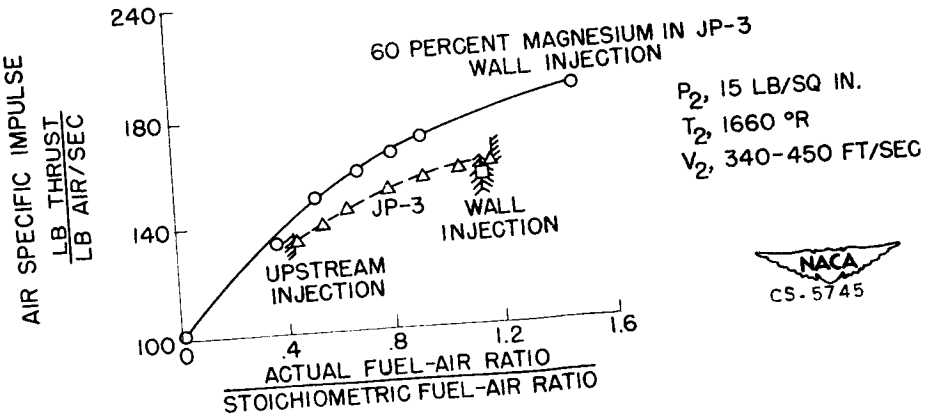


Figure 12

[redacted]

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2946-D

MAGNESIUM SLURRY PERFORMANCE

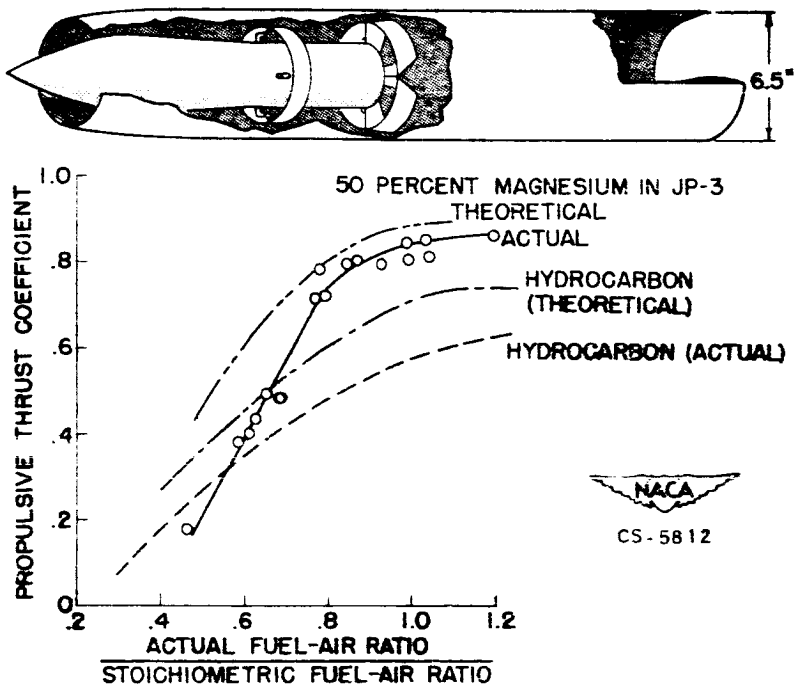

Figure 13

CONFIDENTIAL

DECLASSIFIED

5. COMMENTS Ic

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5. - COMMENTS Ic

By E. J. Manganiello

In the Analysis of Missile Propulsion Systems, the turbojet engine was shown to be competitive for long-range missile application and emphasis was placed on its very desirable low-speed thrust characteristics which significantly reduced the boost requirement for missile acceleration to design flight conditions. It may be of interest, at this time, to consider the detailed turbojet engine design characteristics used to obtain the missile performance presented in the analysis in order to assess the degree of optimism or, alternatively, conservatism involved.

2546-E

The assumed design characteristics are shown in figure 1 along with a schematic representation of the engine. A conventional subsonic axial-flow compressor of nine stages having a maximum efficiency of 85 percent is used. It develops a pressure ratio of 5 at sea-level take-off conditions and has a corrected air flow capacity of 27 pounds per second per square foot of inlet area. The weight flow is corrected by use of the familiar θ and δ factors, where θ is the ratio of total temperature at the compressor inlet to standard sea-level temperature, and δ is the ratio of total pressure at the compressor inlet to standard sea-level pressure. At Mach number 2.5 and altitudes above the tropopause, the compressor pressure ratio is reduced to 2.6 and the corrected air flow, to 19 pounds per second per square foot, or 70 percent of the take-off value. These reductions are the result of the high inlet temperature associated with the high flight speed.

The primary combustor provides a turbine-inlet gas temperature of 2000° R at a combustor efficiency of 95 percent. The turbine efficiency is 85 percent. A conventional V-gutter-type afterburner with a combustion efficiency of 85 percent is used to provide afterburner gas temperatures of 3000° R and higher. The specific weight of the engine is 0.36 pound per pound of thrust. This value is based on the dry thrust of the engine, that is, with the afterburner inoperative, but includes the weight of the afterburner installation.

The foregoing design characteristics or assumptions are considered to be fairly representative of current turbojet engine development. The question is raised whether this level of performance is the limit, or whether significant improvements in performance and operating characteristics can be expected. The next paper presents, from a research viewpoint, some possibilities for making gains in the performance of the turbojet engine for supersonic missile application.

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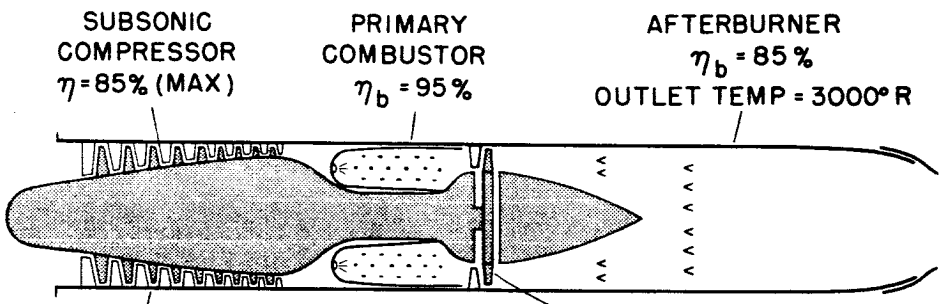
CONVENTIONAL TURBOJET FOR SUPERSONIC PROPULSION



ENGINE SPECIFIC WEIGHT = 0.36

Figure 1

[redacted]ENTIAL

UNCLASSIFIED

6. THE TURBOJET ENGINE FOR MISSILE PROPULSION

By Bruce T. Lundin and Robert O. Bullock

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6. - THE TURBOJET ENGINE FOR MISSILE PROPULSION

By Bruce T. Lundin and Robert O. Bullock

INTRODUCTION

The analysis of missile propulsion systems presented in the first paper has indicated that the turbojet engine is a very effective power plant for moderate-range missiles, and several optimum design and operating conditions of an engine for this application were evaluated. The basic characteristics of the engine considered in this analysis were, in general, typical of present designs and similar to engines in current use in subsonic aircraft. Further improvements in the characteristics of the turbojet engine are, of course, the continuing goal of many research activities and some recent accomplishments of the Lewis laboratory in several fields of work that offer possibilities of markedly increasing the performance capabilities of the turbojet engine for supersonic missile propulsion are summarized herein.

ENGINE OPERATION AT SUPERSONIC FLIGHT SPEEDS

One of the most valuable characteristics of the turbojet engine for missile propulsion, particularly for moderate flight ranges where total missile plus booster weight is considered of primary importance, is the ability of this engine to accelerate the missile from subsonic to supersonic flight speeds without the assistance of large amounts of rocket boost. In order to fully realize this inherent advantage of the turbojet engine, it must operate over a wide range of flight conditions or, more specifically, a wide range of inlet-air temperatures. Operation of current turbojet engines at high inlet-air temperatures is, of course, accompanied by large reductions in compressor equivalent speeds and hence in the mass-flow capacity of the engine. It is this reduction in engine-flow capacity at high inlet-air temperatures that constitutes one of the principal obstacles to the attainment of maximum engine performance for all missile flight regimes.

This problem of engine operation over a wide range of flight conditions is illustrated with reference to the plot of compressor characteristics shown in figure 1. In this figure, the characteristics of the NACA 10-stage axial-flow compressor, which may be considered representative of modern axial-flow compressor designs, are presented as a conventional plot of over-all pressure ratio against specific air flow in pounds per second per square foot of frontal area. Performance is shown for a range of equivalent speeds from 100 percent to 60 percent of the rated value and contours for adiabatic efficiencies of 83, 82, and 80 percent are included.

[redacted]

These characteristics represent the performance of the compressor operating as an isolated component. The operating conditions of a compressor that is an integral component of an engine depends on the flow relations that satisfy the conditions of work and flow continuity in the turbine; these operating conditions may, for an engine operating at constant rotative speed and turbine-inlet temperature, be represented on this plot as a single curve. For the present illustrative example, this equilibrium operating line is drawn through a pressure ratio of 7 at rated equivalent speed. As long as the engine is operating at constant speed and turbine gas temperature, it will therefore operate along this curve at a location established by the inlet-air temperature.

At a flight Mach number of 2.5, the inlet-air temperature is over 400° F, at which condition the value of θ is increased sufficiently to reduce the compressor equivalent speed to 77 percent of its rated value. Operation at the supersonic cruise condition therefore occurs at the location indicated by the cross symbol. At this point, the compressor pressure ratio is reduced to nearly half of its rated value and the air flow is reduced to less than 70 percent of the rated value. Although this decrease in pressure ratio with increasing flight speed is consistent with thermodynamic-cycle requirements, the reduction in air flow means that the engine must be simply that much larger and heavier to provide the required thrust.

Methods of engine operation that would maintain the operating point at the design value for all flight speeds, and thus maintain the engine flow capacity always near its maximum value, would therefore greatly increase the performance capabilities of the engine at supersonic flight speeds. This objective may be accomplished by adjusting the actual engine speed N in accordance with the inlet temperature θ to maintain a constant value of the equivalent compressor speed $N/\sqrt{\theta}$. This procedure is, of course, equivalent to designing the engine, particularly the structural and aerodynamic limits of the compressor and turbine, to be matched at the cruise condition rather than at take-off and to operate at a reduced rotative speed at take-off instead of a reduced equivalent speed during the entire cruise flight.

This method of engine operation is compared with the conventional constant-speed method in a somewhat graphic manner in figure 2. In this figure, two engines are shown: the conventional, take-off design engine, and the constant-equivalent-speed engine that will be designated as the supersonic engine. Pertinent engine performance variables at both take-off and cruise are shown adjacent to the engine sketches and the compressor operating point of each engine is illustrated by the small compressor characteristic plots. For the take-off engine, the actual, or structurally limited, speed N and the aerodynamic speed $N/\sqrt{\theta}$ are matched at the take-off condition with limiting

conditions being represented by a value of unity. At this condition, the specific air flow is, for example, equal to 27 pounds per second, the turbine-inlet temperature is 2000° R, and the pressure ratio is set at a value of 5. At the supersonic cruise condition, the rotative speed N remains at a value of unity, but the equivalent speed is, as previously mentioned, reduced to 0.77. As a result, the specific air flow of the engine is only 19 pounds per second and the pressure ratio is reduced to 2.6. The turbine-inlet temperature is maintained at a constant value of 2000° R by appropriate adjustment of the exhaust area.

2546-F

For the supersonic engine, which has been redesigned aerodynamically to handle a greater mass flow at higher inlet-air temperatures, the actual rotative speed N and the equivalent speed $N/\sqrt{\theta}$ are matched at the cruise condition and both take-off and cruise occur at the same compressor operating point. This method of operation requires that the engine speed be reduced at take-off, but both the air flow and the pressure ratio are constant. The pressure ratio in this present example was reduced to a value of 4 as a compromise between optimum values at both take-off and cruise. Although it would be possible to operate the engine at a rotative speed greater than 0.77 of the rated value at take-off, such operation would increase the aerodynamic speed $N\sqrt{\theta}$ beyond the usual design values and would, in general, be accompanied by both sufficient choking of the flow in the first stages of the compressor and by reductions in compressor efficiency to preclude any significant increase in engine thrust. Because both the pressure ratio and air flow of this engine are constant, flow continuity in the turbine requires that the temperature ratio between the turbine inlet and the compressor inlet be constant, with the result that the turbine-inlet temperature at take-off is reduced to 1180° R. Attempts to increase this temperature would, of course, increase the pressure ratio of the compressor and drive the compressor into surge.

A comparison of the thrust capabilities of the take-off and supersonic engine is shown in figure 3, where a thrust coefficient, which is proportional to the corrected net thrust per unit of nacelle frontal area, is plotted against the flight Mach number. A curve of missile drive coefficient is included for comparison. Variations in altitude for the various flight Mach numbers, as prescribed for a typical flight plan, were integrated into the calculations for these curves; the increase in altitude from sea level to 35,000 feet at a flight Mach number of 0.85 was responsible for the discontinuity in the curve for the take-off engine. At a cruise Mach number of 2.5, the supersonic engine is capable of producing a thrust that is about 25 percent greater than that of the take-off engine because of its higher air flow capacity. This larger thrust coefficient of the supersonic engine would permit the use of smaller, lighter engines and be reflected in a

10 to 15 percent decrease in total weight for a moderate-range missile. At take-off and low flight speeds, however, the thrust of the supersonic engine is only about one-quarter that of the take-off engine, because of the reduced turbine-inlet temperature at these flight conditions. The adverse effect of this loss in engine performance at low speeds is apparent by comparison of the thrust capabilities of the engine with the missile drag characteristics. For the example of figure 3, the thrust is less than the drag throughout most of the range of flight speeds and the missile would have to be boosted by auxiliary rockets to a flight Mach number of over 2. Thus, in order to utilize the superior cruise performance of the supersonic engine at high flight speeds, it would be necessary to employ sufficient rocket boost to accelerate the missile to nearly full flight speed and one of the outstanding advantages of the turbojet engine would be lost.

2546-F

The possibility of overcoming this low thrust at take-off by means of thrust augmentation is illustrated in figure 4. In this figure, both the thrust and drag coefficients are plotted against flight speed as before, except that in this case the missile drag has been adjusted to match the thrust of the engine at the cruising flight speed. The amount that the engine thrust may be increased by known methods of thrust augmentation is represented by the shaded area above the basic thrust curve. At take-off conditions, the pressure ratio across the turbine is very high in order to produce the required power with the low turbine-inlet temperature. As a result, the pressure in the tail pipe of the engine is so low that only a very small amount of afterburning is possible. Therefore, although the available thrust augmentation increases very rapidly as the flight Mach number is increased, the missile would still have to be boosted by auxiliary rockets to supersonic speeds.

Significant increases in the basic thrust of the supersonic engine at low flight speeds are therefore necessary if this engine is to be developed into an effective missile power plant. Such thrust increases will obviously require the utilization of higher turbine-inlet temperatures without the occurrence of compressor surge. One method of accomplishing this objective is to provide the engine with a variable-area turbine, in which case the turbine area could be increased sufficiently at take-off to permit the use of higher gas temperatures at the same condition of compressor pressure ratio and air flow. The use of a variable-area turbine nozzle would, in addition to simply permitting a higher turbine-inlet temperature at take-off, provide the engine with an additional degree of freedom and attendant flexibility in the selection of optimum operating conditions of pressure ratio, efficiency, and air flow over the wide ranges of conditions encountered in high-speed missile applications.

With this added degree of flexibility in the selection of the engine operating point, the optimum method of engine operation would

depend, to a large degree, on how much the pressure ratio may be reduced as the flight speed is increased, or, in other words, how closely it may be permitted to approach optimum values without being adversely affected by reductions in compressor efficiency or air flow. Without recourse to the detailed calculations necessary to the establishment of such optimum conditions for various specific engines, a somewhat simpler and more generally applicable case of an engine operating at a fixed compressor point was computed and the thrust capabilities of this engine are presented in figure 5 in which the thrust coefficient is plotted against the flight Mach number. Curves are shown for the conventional take-off engine, the fixed-area supersonic engine, and the supersonic engine with a variable-area turbine nozzle. With the variable-area turbine, the thrust of the supersonic engine may be increased to a value equal to that of the take-off engine at take-off conditions. At all other flight speeds, the thrust of this engine is greater than that of the conventional engine because of its higher air-flow capacity until, at a Mach number of 2.5, the 25 percent greater thrust coefficient is obtained. The thrust of this supersonic engine may furthermore be effectually increased by various methods of thrust augmentation that are to be discussed subsequently herein, in order to achieve still further improvement in the take-off and acceleration characteristics of the missile.

2546-F

The use of a variable-area turbine to increase the flexibility of engine operation has, of course, long been recognized, but formidable problems of both mechanical design and performance have usually been anticipated. Some performance data have, however, been obtained recently on a variable-area turbine provided by an engine manufacturer as a special component of a full-scale engine. This variable-area turbine nozzle was not designed as a regular component of the engine, and high performance was not a primary objective; the nozzle was intended solely as a means of regulating the engine operating point for other component investigations. In the design used, the blades of the first-stage nozzle assembly of a two-stage turbine were mounted on small shafts or pins and the flow area was varied by merely rotating the blades in unison by means of a simple linkage system. This turbine was in use for over 100 hours of engine operation and proved to be entirely satisfactory from a mechanical standpoint.

The measured efficiency and the flow capacity of this turbine operating as an integral part of the engine are shown in figure 6, in which the turbine adiabatic efficiency and the gas flow corrected to turbine-inlet conditions are plotted against the actual turbine-nozzle area in square feet. Data are shown for engine speeds of 100, 94 and 87 percent of the rated value. The range of area adjustment provided by this turbine was from about 1.1 to 1.7 square feet. For flow areas greater than about 1.4 square feet, however, the flow coefficient decreased in such a manner that the gas flow remained nearly constant,

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with the result that the effective change in area as determined by the change in turbine gas flow was of the order of 12 to 15 percent. As previously mentioned, however, this variation in flow capacity was obtained by modulation of only the first stage of a two-stage turbine and greater variations could undoubtedly have been obtained if either the area of both stages had been varied or if a single-stage turbine had been used. The turbine efficiency, shown in the upper part of the figure, is seen to be nearly constant for all engine speeds over the range of effective areas investigated and has a value of 85 percent at rated engine speed. It thus appears possible, by means of a relatively simple and reliable variable-area turbine-nozzle assembly, to incorporate into the engine sufficient fluid control without loss in efficiency to permit considerable freedom in the selection of optimum engine operating conditions.

2546-5

TRANSONIC COMPRESSOR

In the preceding section some of the engine operating problems arising from the wide range of inlet temperatures encountered in a missile application were reviewed and the possibility of considerably improving the engine performance capabilities at high flight speeds and still retaining good take off or low speed performance by the application of a variable-area turbine nozzle and proper engine speed control were outlined. These considerations have, of course, been based on engine components of conventional design. It therefore becomes pertinent to examine the possibilities of still further increasing the effectiveness of the engine by utilization of different or more advanced types of components.

Research efforts to further improve the performance capabilities of the turbojet engine are naturally directed, to a large degree, toward methods of increasing the air flow capacity and of decreasing the weight of the engine. Direct reductions in engine weight may be accomplished by increasing the pressure ratio per stage of the axial-flow compressor because a smaller number of stages would then be required to obtain a given over-all pressure ratio. A reduction in the number of stages might also be considered as a means of improving compressor range and efficiency by reducing the stage matching difficulties generally encountered with large numbers of stages.

In general, an increase in both compressor mass flow and stage pressure ratio can be achieved by increasing the axial Mach numbers at the inlet to the first rotor row. Increased relative inlet Mach numbers can also permit the selection of wheel speeds in accordance, primarily, with compressor and turbine stress limitations and allow greater freedom in velocity diagram design. In any event, substantial improvements in engine performance may be achieved by extension of allowable relative inlet Mach numbers beyond present limiting values.

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Results of recent investigations of supersonic axial-flow rotors have indicated acceptable performance when these units are operated at transonic relative inlet Mach numbers; the concept of utilizing a transonic stage as the first stage of an axial-flow compressor is thus introduced. The utilization of higher relative inlet Mach numbers at higher compressor speeds would also greatly increase the effectiveness of all the following stages of the compressor, because the temperatures are normally higher in this region and the Mach number limits are accordingly reduced.

2546-F

Within recent months, an exploratory axial-flow-compressor inlet stage designed specifically to operate in the transonic range of inlet Mach numbers has been constructed and tested at the Lewis laboratory and some data on its performance characteristics are now available. This transonic inlet stage contained no inlet guide vanes and was designed for relative rotor inlet Mach numbers ranging from 0.75 at the hub to 1.13 at the tip. The stators were designed for conventional values of stator-inlet Mach number. A photograph of this rotor is shown in figure 7. The rotor is 17.36 inches in diameter and contains 21 blades having a hub-to-tip ratio of 0.525 at the inlet and 0.6 at the outlet. The blade depth is about 3 inches and the thickness varies from about 6 percent at the tip to 8 percent at the hub. The blade shape at the tip is very similar to a symmetrical, circular-arc profile.

This compressor stage was operated in a variable component test rig and some of the data obtained are presented in figure 8 where the overall adiabatic efficiency and pressure ratio are plotted against the corrected weight flow per unit frontal area. The solid curves in this figure are for the transonic compressor and the dashed curves represent comparative performance of a more conventional subsonic inlet stage of a research compressor having a high blade loading. Performance is shown for rotor speeds of 100 percent and 60 percent of the design value for the transonic compressor and for 100 percent and 50 percent of the design speed for the more conventional stage.

At the design speed, the transonic compressor had a peak efficiency of 90 percent and developed a pressure ratio of 1.47 at an equivalent air flow of nearly 30 pounds per second per square foot of frontal area. This pressure ratio is equivalent to more than two conventional subsonic stages and would thus significantly reduce the total number of stages required because of this higher pressure ratio itself and because the higher wheel speeds permitted by this design would increase the effectiveness of all the succeeding compressor stages.

As illustrated in figure 8, the peak efficiency of the transonic stage remained at 90 percent throughout the range of speeds investigated. This good efficiency is also maintained over a much larger range of flow rates than is usually possible with conventional designs, particularly

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at the lower speeds, which may be expected to contribute to good compressor range when this stage is used as the inlet of a multistage compressor. This wide range of the stage is due largely to the elimination of inlet guide vanes, and the attendant relatively small change in angle of attack for a given change in air flow.

The application of this transonic stage as the inlet of a multistage compressor requires not only that it have acceptable performance itself, but that the stator outlet conditions be satisfactory for staging with conventional subsonic designs. Some of the flow conditions at the stator outlet are presented in figure 9 where the Mach number distribution and the local adiabatic efficiency are plotted against the radius ratio from hub to tip. The local adiabatic efficiency of the transonic stage, which is a direct measure of the quality of the flow, has roughly the same distribution across the blade passage as the conventional subsonic stage. Stator-outlet Mach numbers, although exhibiting a somewhat greater variation across the blade passage than with the conventional design, do vary in a regular manner across the outlet radius and the highest value measured was less than 0.60. The stator-outlet flow angles also showed little change across the blade height and varied only 3° over the range of speeds investigated. For the measured outlet angles and Mach numbers, all Mach numbers relative to a succeeding rotor row would fall within current limits. Stator-outlet conditions would thus seem to be favorable for the matching of succeeding stages of current design.

2546

TURBINE COOLING

The very satisfactory performance obtained with this initial, exploratory transonic rotor has encouraged continuing research in this field and more advanced designs to operate at higher speeds and having greater flow capacity are now being actively pursued. With the successful development of these new designs, the principal problem of handling higher mass flows at higher rotative speeds will be determined to a large extent by turbine-blade stresses rather than by compressor aerodynamics. Means of raising the stress limits of turbine blades therefore become necessary if the ultimate utilization of these advanced types of compressor is to be achieved. One very promising method of permitting higher turbine-blade stresses is to reduce the metal temperatures by turbine-blade cooling. When it is recalled that a reduction in blade temperature of about 400° will double the strength of many current blade alloys, the powerful effect of turbine-blade cooling is realized. Effective turbine-blade cooling will also permit use of higher turbine-inlet gas temperatures, the advantages of which have been pointed out in the paper entitled "Analysis of Missile Propulsion Systems." Methods of turbine-blade cooling have been the subject of considerable research effort at the Lewis laboratory and significant results of this work have been summarized from time to time at various

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conferences and in numerous publications. Some preliminary heat-transfer investigations have recently been completed on a new type of air-cooled blade that is reviewed with reference to figure 10. In this figure, the effectiveness of the blade-cooling method is presented as a plot of blade temperature against cooling-air flow for a given gas temperature and cooling-air temperature.

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The primary objective of most of the turbine-blade cooling research has been to decrease the cooling-air flow requirements and to obtain effective cooling of the critical leading and trailing edges of the blades. An important factor in reducing the cooling-air requirements is the design of the internal cooling passages. In designs previously reported, such as illustrated by the internal tube blade (fig. 10), extended heat-transfer surfaces were inserted in the form of fins or hollow tubes within a shell that formed the blade profile. In such a design, the hollow shell acts as the load-carrying element of the blade and is directly exposed to the hot gases. In the type of blade more recently investigated, which is illustrated as an internal-strut blade, a more effective integration of both the cooling and structural characteristics of the blade is accomplished by concentrating the load-carrying element in the central portion of the blade by a strut to which is attached a thin shell that performs the aerodynamic function of the blade. In this manner, the load-carrying structure is almost completely submerged within the cooling air and the thin shell, which is spot-welded to the strut at numerous locations to prevent concentration of centrifugal loads, is not highly stressed and may therefore be operated at higher temperatures.

Based on heat-transfer investigations of these blades, the blade temperatures that would be obtained at a gas temperature of 2500° R and a cooling-air temperature of 1240° R have been computed and are presented by the curves of figure 10. This relatively high cooling-air temperature is the temperature of the air at the compressor outlet of a typical engine at a flight Mach number of 2.5. For the internal strut blade, curves are shown for both the load-carrying strut and for the hotter but lightly loaded shell. Metal temperatures of approximately 1100° F for the strut are possible for a cooling-air flow rate about 3 percent of the engine air flow, even for these rather severe gas and cooling-air temperature conditions. The attainment of these metal temperatures would therefore permit use of low-alloy steels at stress levels of the order of 50,000 pounds per square inch. The shell temperature of this blade is about 200° hotter than that of the internal strut, but because it carries no severe tensile loading, it may be operated at these higher temperatures without failure. Furthermore, if desired, a corrosion-resistant high-temperature metal shell may be used with a non-critical metal strut, providing a substantial saving in over-all usage of critical material when compared with a blade made entirely from critical alloys. Comparative blade temperatures of the load-carrying

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shell of the internally tubed blade are even higher than the shell temperature of the strut-supported blade and are of such a magnitude as to obviate their application for these severe cooling requirements.

THRUST AUGMENTATION

In the foregoing discussion, it has been observed that the utilization of the transonic-type compressor and an air-cooled turbine blade, together with the added flexibility provided by a variable-area turbine nozzle, offer attractive possibilities for greatly increasing the effectiveness of the turbojet engine and maintaining good thrust characteristics both at low flight speeds and at supersonic cruise. In addition to the need for an engine that is capable of subsonic flight acceleration, the boosting or take-off of the missile from rest on the ground to a practical flying speed is of equal importance to the overall missile propulsion requirements. Regardless of the method of take-off that may be used, special means of increasing the thrust of the engine for short periods of time by thrust augmentation methods will be of value in either minimizing the amount of rocket booster required or in reducing the take-off distance to practical, minimum values.

Methods of achieving maximum possible thrust augmentation for either take-off or transonic-flight use have been the objective of several experimental investigations and some of the more significant results are summarized in figures 11 to 13. This discussion will be devoted first to thrust augmentation by various methods of water injection, and later to the combination of water-injection and afterburning.

In figure 11 are summarized some recent water-injection results at sea-level static conditions in a plot of augmented thrust ratio, or ratio of augmented thrust to normal engine thrust, against the injected water-alcohol to air ratio. About 30 percent of alcohol was included in the injected mixture to provide the extra fuel required during injection. Two sets of data are included in this figure; the lower curve for injection into the combustion chambers, and the upper set of curves for the combination of combustion-chamber injection and compressor interstage injection.

For injection into the combustion chambers, the mixture of water and alcohol was injected into each combustor of the engine by four spray nozzles that were located approximately in the middle of the combustor and that directed the liquid into the primary burning zone. Interstage injection was accomplished by injecting the liquid into the sixth stage of a 12-stage compressor by means of 18 small tubes that extended approximately halfway across the annular passage of the compressor and that were fitted with small, high-penetration-type nozzles.

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For injection into the combustion chamber, the thrust increased linearly with injection rate and, for this engine, reached an augmentation ratio of 1.25 at a liquid-to-air ratio of about 0.1. Because the injection of a liquid into the combustion chambers shifts the operating point of the compressor to higher pressure ratios, the maximum amount of thrust augmentation that may be obtained for any particular engine will depend on the surge characteristics of the compressor. For these data, the pressure ratio of the compressor increased from a normal value of 4.7 to a maximum of 5.1, or an increase of 8 percent. The line that is drawn through the data points is a calculated thrust augmentation based on the normal compressor map for the engine. The excellent agreement of this calculated line with the experimental data illustrates that the application of this method of thrust augmentation may be accurately predicted for any particular type of engine. It is also of interest to note that, even for this liquid flow of as much as 10 percent of the engine air flow, no adverse effects on the combustion processes other than a moderate decrease in efficiency at the higher injection rates were experienced and no measurable shift in the gas-temperature profile occurred.

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The use of interstage injection, which was developed at the Lewis laboratory several years ago in an effort to overcome centrifugal separation of the injected liquid and air normally experienced in axial-flow compressors, is shown by the dash line in the upper part of figure 11 and is seen to be somewhat superior to combustion-chamber injection over most of the range of liquid flows. Recent data on the combination of interstage injection and combustion-chamber injection are represented by the data points. For these runs, the amount of interstage injection was held constant at a liquid-air ratio of about 0.02 and the amount of injection into the combustor was continually increased. A linear increase in thrust augmentation with injected flow is also apparent here, with a maximum augmentation ratio of 1.4 being reached at a liquid-to-air ratio of 0.14. Thermodynamically, these two methods may be expected to augment each other and, in fact, the line through the data points may be closely approximated by the product of the two augmentation methods separately.

This maximum augmentation ratio of 1.4 is, of course, slightly less than values usually obtainable by afterburning and the liquid consumption is about three times as great. The choice of this method of augmentation in preference to afterburning must therefore be made on the basis of short-duration application where simplicity of installation is of primary importance. A further outstanding application of this method of thrust augmentation is its use in combination with afterburning when very large thrust increases are required. Several such investigations of the combination of water injection and afterburning have been conducted and, while total thrust ratios of about 1.7

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have been achieved, the full potentialities of this method have not yet been realized because of difficulties of maintaining satisfactory or efficient combustion in the afterburner at high rates of water injection. This difficulty has led to investigations into the use of high-energy metallic fuels in the afterburner because of their well-known superior combustion characteristics.

While present limitations to the availability of these high-energy fuels have limited full-scale engine tests to brief exploratory runs, a series of experiments has been recently completed with a 6-inch-diameter combustor that closely simulated the operating conditions in an engine tail pipe. Some of the data obtained with this combustor are presented in figure 12 where the water-air ratio that was injected upstream of the combustor is plotted against the afterburner equivalence ratio, or the ratio of the fuel-air ratio of the particular run to the stoichiometric mixture for the fuel. These data illustrate, primarily, the limits of stable combustion in the presence of water, with the circles representing standard JP-3 fuel and the squares representing a slurry of 60 percent magnesium in a JP-3 carrier. The solid or closed circles indicate combustion blow-out.

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For the conventional JP-3 fuel, the limits of stable combustion were gradually decreased as the water-air ratio increased; operation at water-air ratios in excess of 0.08 was impossible. These limits agree well with the operating limits experienced with a full-scale afterburner. With the 60 percent magnesium slurry fuel, satisfactory operation was achieved throughout the range investigated with maximum water-air ratios being limited to from 0.16 to 0.18 because of pumping limitations of the equipment used rather than combustion characteristics. Although not illustrated by these data, the combustion efficiency with the magnesium slurry fuel was also unaffected by the amount of water injected, except at the very highest water-air ratios. Satisfactory operation of an afterburner with magnesium slurry fuel at stoichiometric fuel-air ratios and for water-air ratios in the region of 0.14 to 0.16 is therefore indicated.

While, as previously mentioned, data for combined thrust augmentation by water injection and a magnesium fueled afterburner are not available, sufficient data do exist upon which to base a fairly accurate estimate of engine performance and some of these data and predictions are presented in figure 13. In this figure, the augmented thrust ratio for take-off conditions is plotted against the augmented liquid ratio, which is defined as the ratio of the total liquid flow through the engine in the augmented condition to the normal engine fuel flow. Curves are presented for water injection alone, for afterburning alone, and for the combination of afterburning and water injection. Afterburning performance is shown for both JP-3 fuel, as obtained from full-scale engine tests,

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and for magnesium slurry fuels as obtained with the 6-inch combustor. A comparative curve for thrust boost by rocket assist is also included.

For afterburning alone with JP-3 fuel, rather familiar performance data are indicated; a thrust ratio of slightly over 1.5 was obtained for an augmented liquid ratio of about 4. For the combination of afterburning and water injection, a maximum thrust ratio of about 1.7 was obtained, as was mentioned previously. During these runs, the rate of water injection was low enough to avoid a decrease in afterburner combustion efficiency and, of particular importance to the present discussion, the curve through the data points was equal to the product of the two augmentation methods separately. It is of interest to note, therefore, that the combined thrust augmentation obtained by interstage injection, by combustion-chamber injection, and by afterburning will closely approximate the product of the augmentation produced by each method separately, provided, of course, that the combustion efficiency is not seriously reduced. The use of afterburning alone with a magnesium slurry fuel is represented by the diamond symbols; somewhat greater thrust increases were obtained than for the JP-3 fuel because of the higher gas temperatures and the greater fuel weight at stoichiometric mixture.

With this establishment of the manner in which water injection and afterburning combine and the possibility of satisfactory combustion with magnesium fuel, the combined use of such an augmentation method may be estimated by simply computing the product of the water injection curve with any given thrust augmentation point on the afterburner curve. This computation has been made for the maximum thrust point of the magnesium fueled afterburner and is presented by the dot and dash curve. By this method then, an augmented thrust ratio of 2.3, or a thrust increase of 130 percent, is estimated for an augmented liquid ratio of about 14. Comparison with the curve for rocket assist shows that this method of engine-thrust augmentation will provide about 60 percent greater thrust for the same investment in liquid or fuel consumption.

CONCLUSIONS

Recent results in several fields of engine research have been reviewed that offer promise of considerably increasing the value and effectiveness of the turbojet engine as a missile power plant. This research has admittedly not yet progressed to the point of practical application or integration into complete engines - and, indeed, much of it is still in rather preliminary stages - but nonetheless it clearly indicates the promise of continued future advancements in engine design and performance. Some of these research achievements are, by way of summary, highlighted and compared with a current, conventional engine in figure 14. For the conventional engine, shown

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in the upper part of the figure, a subsonic compressor is indicated with a pressure ratio of 5 being achieved in nine stages of compression and for an air flow of 27 pounds per second. A turbine-inlet temperature of 2000° and an afterburner operating at an outlet temperature of 3000° are also indicated.

For what has been designated as a supersonic engine, the same thrust output may be obtained for an engine 25 percent smaller in size by maintaining maximum flow capacity at high inlet-air temperatures through the use of a simple variable-area turbine nozzle and proper engine-speed control. Further improvements in engine performance are also promised by recent transonic-compressor research that will enable the inlet stage of the compressor to operate at relative inlet Mach numbers much greater than current subsonic design limitations and hence permit both a greater air-flow capacity and a greater pressure ratio across each stage of the compressor. For an over-all pressure ratio of 4, a five-stage compressor that can handle about 30 pounds of air per second per square foot of frontal area is conservatively estimated.

The higher air flows and wheel speeds provided by a transonic compressor will further aggravate the difficulties of turbine-blade stresses and relief may be found here in turbine-blade cooling. Heat-transfer data on an internal strut blade in which the main load-carrying element is submerged in cooling air were reviewed that indicated satisfactory cooling at a gas temperature of 2500° R for the high cooling-air temperatures associated with supersonic flight. For this engine, therefore, the use of a higher turbine-inlet temperature during cruise rather than an afterburner operating at 3000° R is suggested to further reduce engine size and weight.

In addition to these features of engine design that are aimed at improving the performance during supersonic acceleration and cruise, short-duration thrust augmentation by means of the combination of interstage injection, combustion-chamber injection, and a magnesium fueled afterburner may increase the engine thrust to 2.3 times its normal value at take-off and thus further improve the take-off and transonic acceleration characteristics of a missile.

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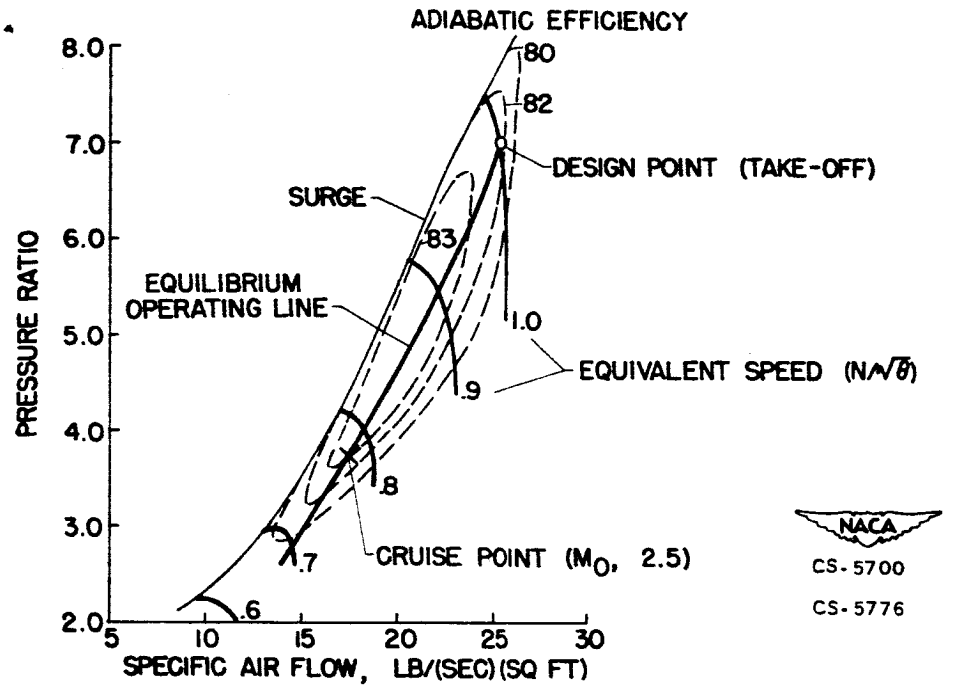

Figure 1

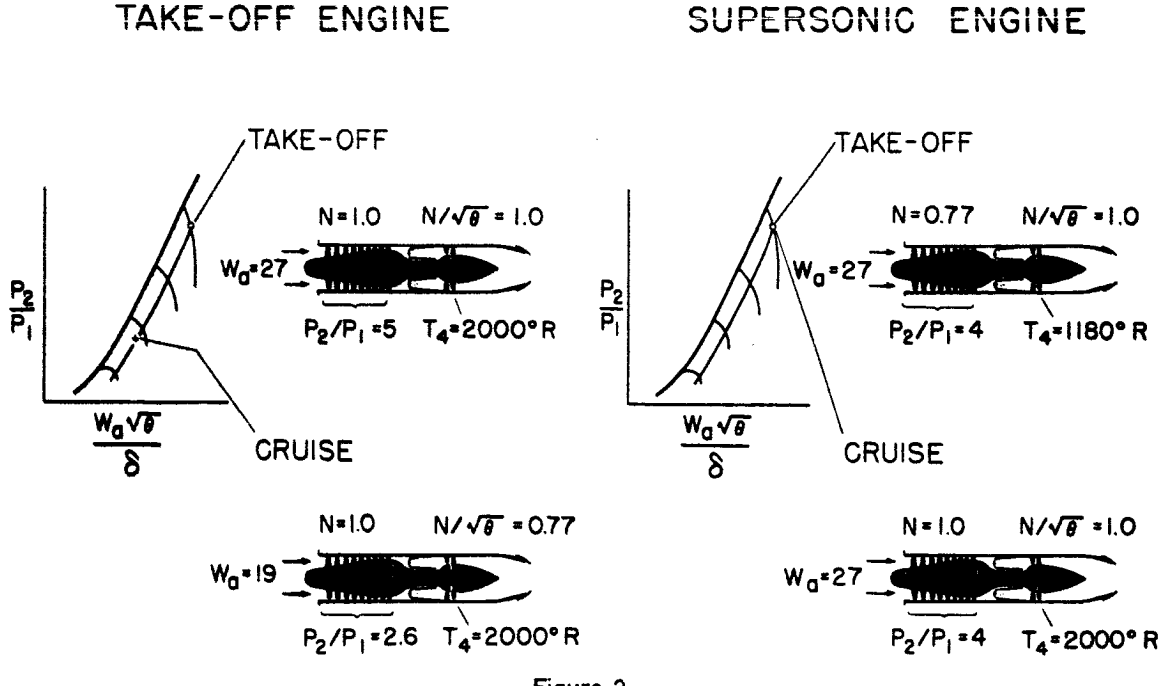

Figure 2

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COMPARISON OF THRUST OF TAKE-OFF AND SUPERSONIC ENGINE WITH MISSILE DRAG

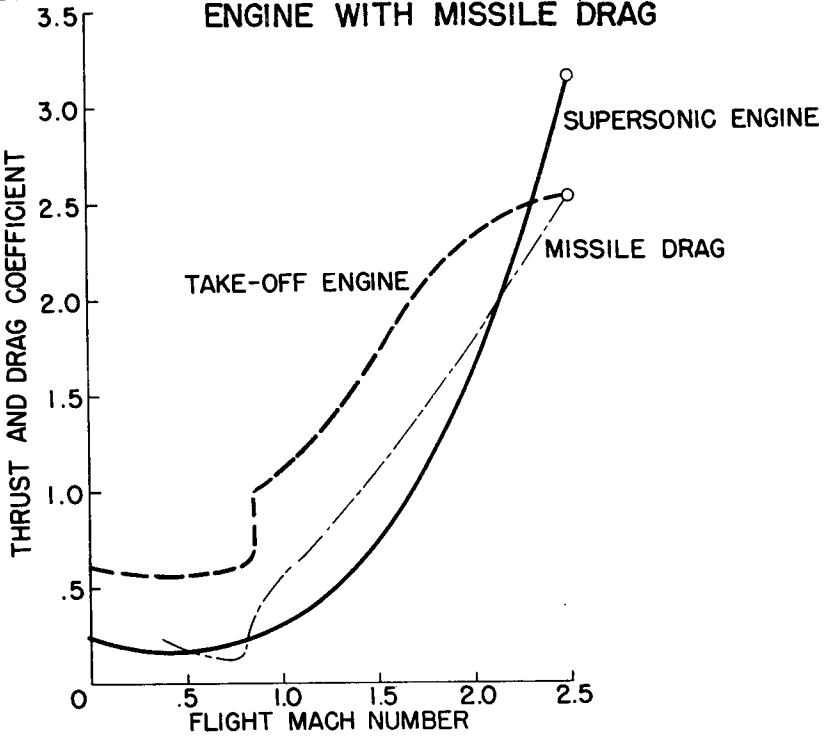

Figure 3

APPLICATION OF THRUST AUGMENTATION TO SUPERSONIC ENGINE

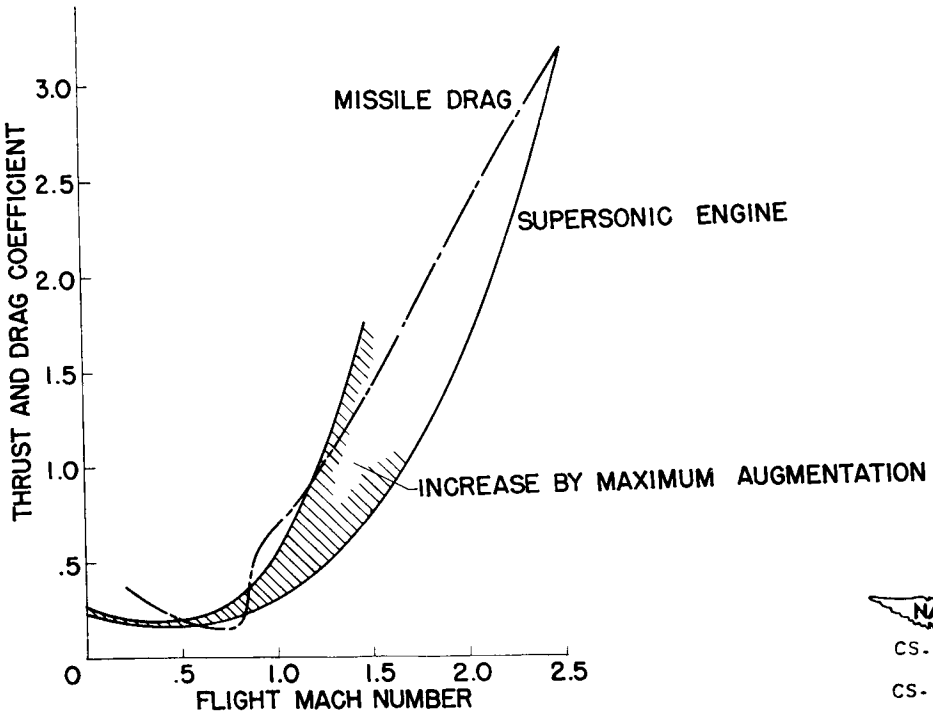

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Figure 4

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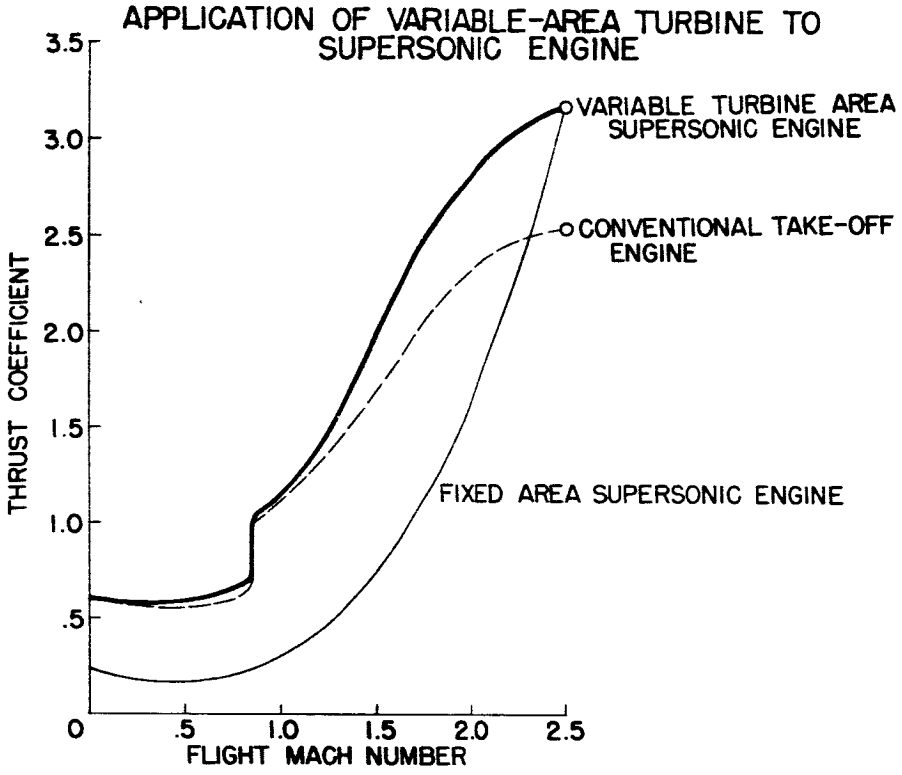

Figure 5

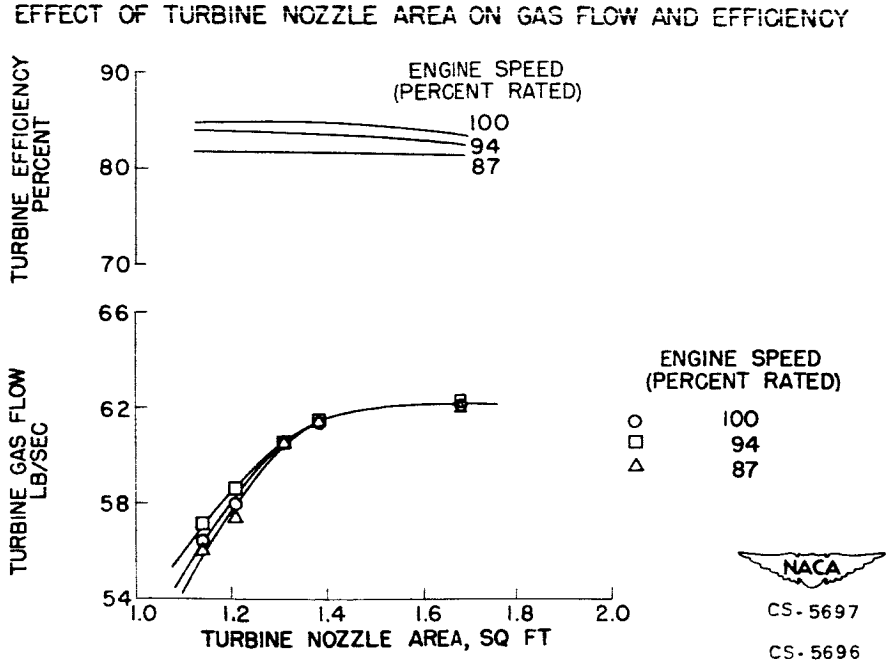

Figure 6

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TRANSONIC COMPRESSOR ROTOR

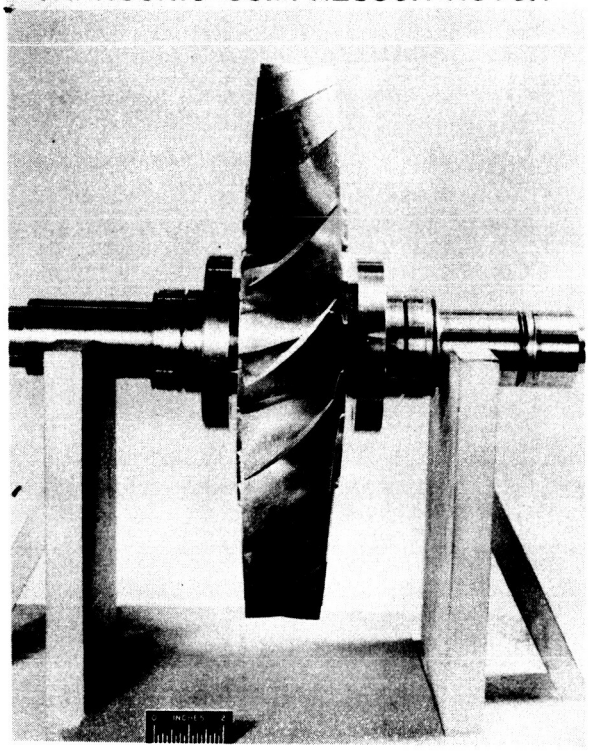

Figure 7

PERFORMANCE CHARACTERISTICS OF TRANSONIC INLET STAGE

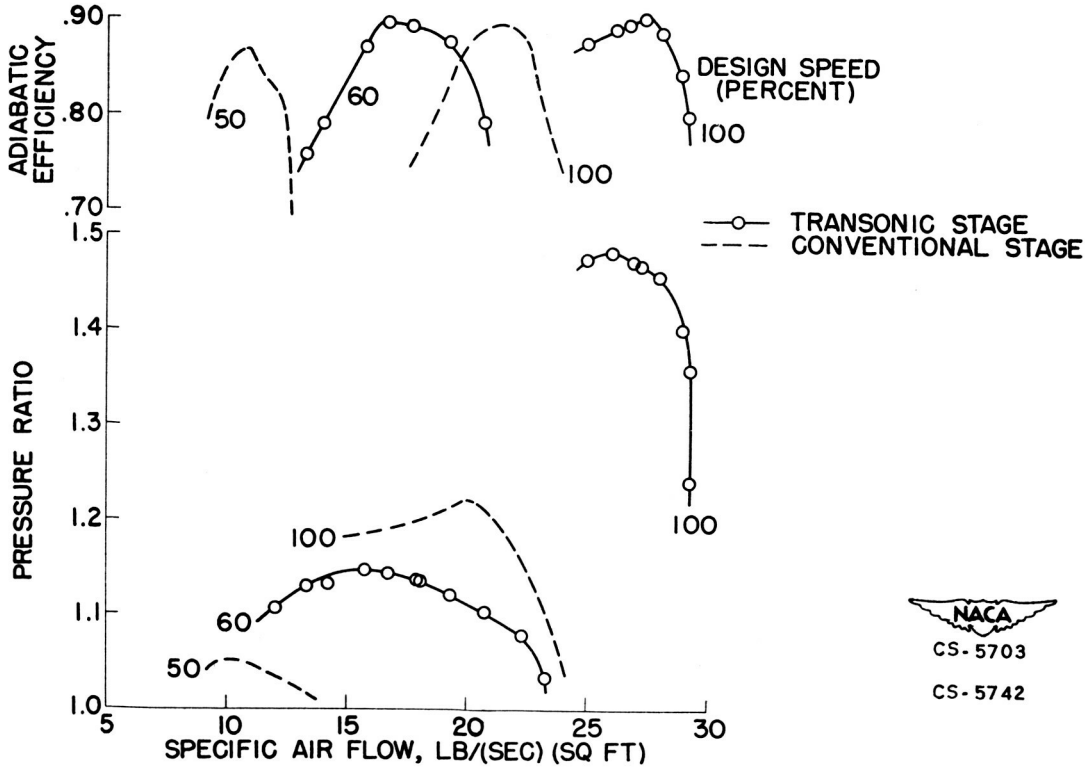

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Figure 8

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FLOW CONDITIONS AT STATOR OUTLET OF TRANSONIC INLET STAGE

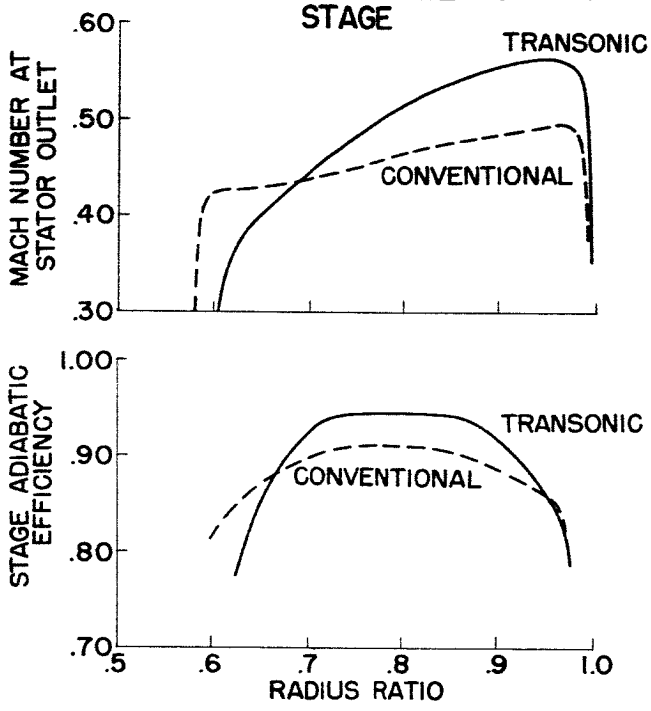

Figure 9

BLADE TEMPERATURES OF AIR COOLED TURBINE BLADES
GAS TEMPERATURE, 2500° R
COOLING-AIR TEMPERATURE, 1240° R

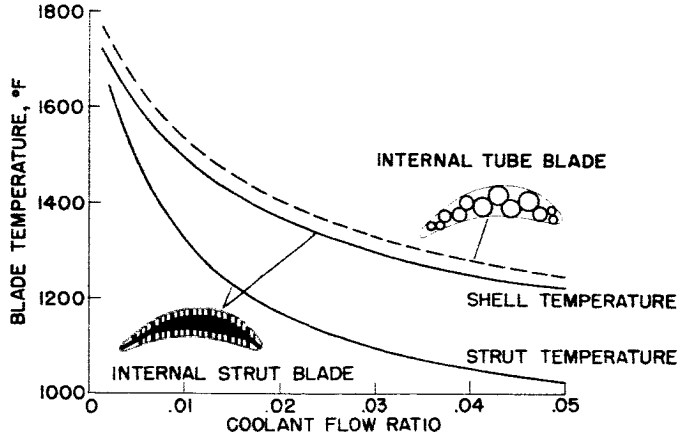

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Figure 10

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THRUST AUGMENTATION BY WATER-ALCOHOL INJECTION

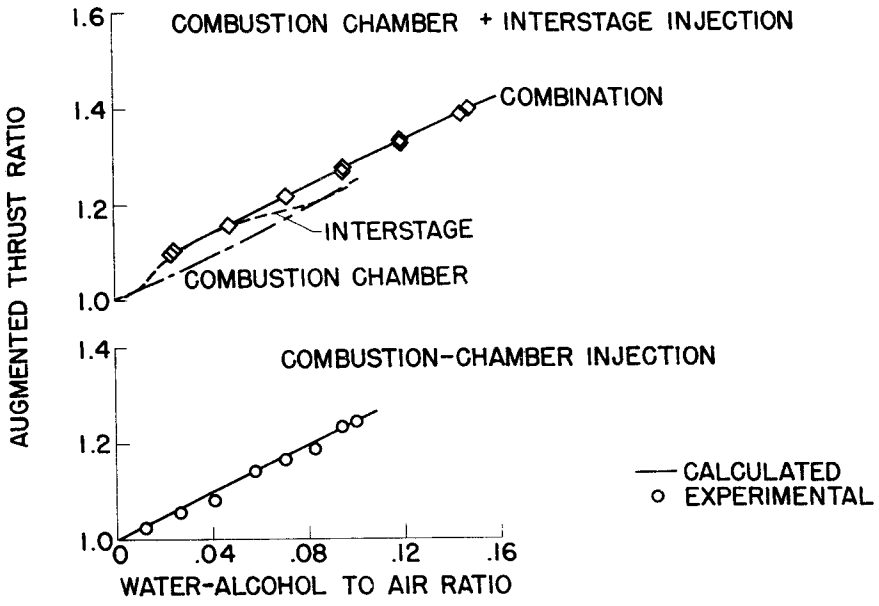

Figure 11

LIMITS OF COMBUSTION WITH WATER INJECTION

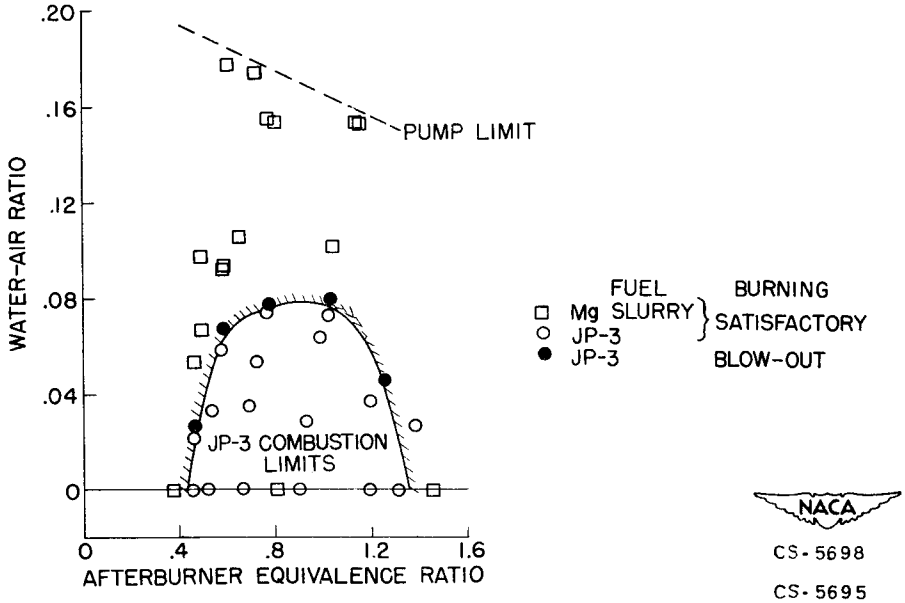

Figure 12

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TURBOJET THRUST AUGMENTATION FOR MISSILE TAKE-OFF

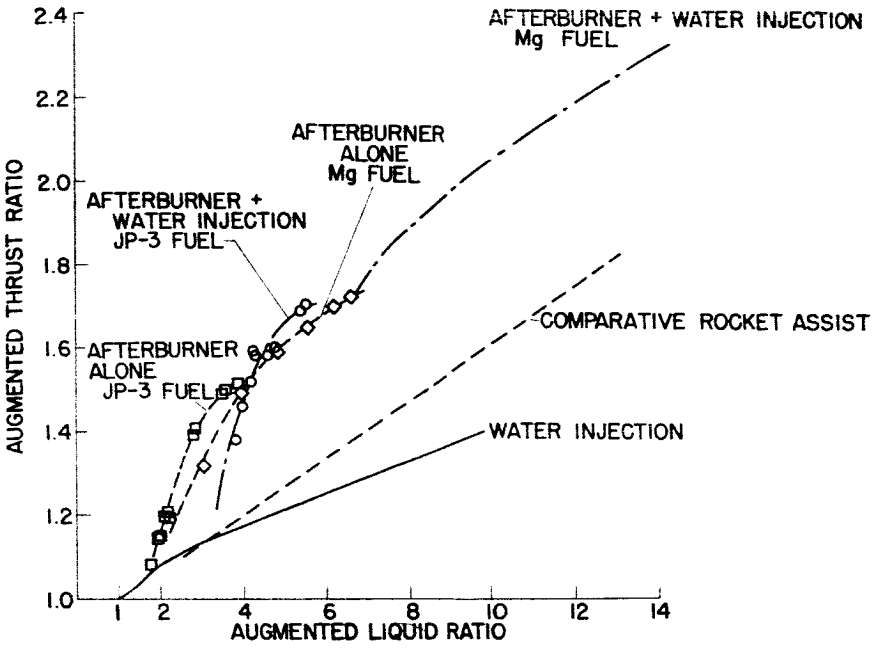
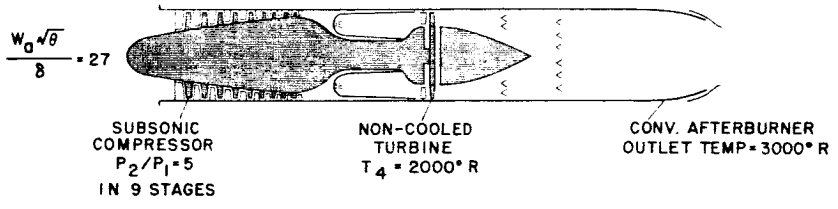


Figure 13

TURBOJET FOR SUPERSONIC PROPULSION

CONVENTIONAL ENGINE



SUPERSONIC ENGINE

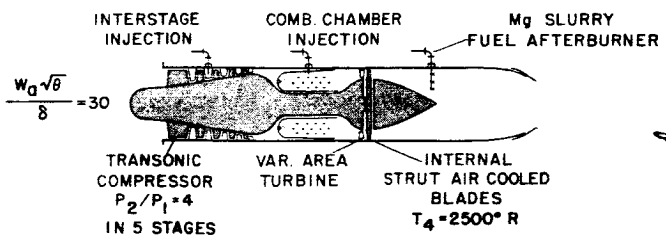


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Figure 14

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7. - COMMENTS Id

By E. J. Manganiello

The rocket was shown to be of interest, not only as one of the competing engine types for the primary propulsive unit of supersonic missiles, but also as a booster for launching missiles powered by other engine types.

In the analysis various ranges of rocket size, impulse, and duration were involved. For example, the rocket booster required for launching and accelerating the 6000-mile ram-jet missile to its design Mach number of 3.5 at 70,000 feet altitude has a total thrust of 240,000 pounds divided between two units. Gasoline and oxygen, having an experimental specific impulse of about 240 seconds, was used as the propellant combination. The gross weight of the rocket unit is about 80,000 pounds and the burning duration is 72 seconds.

For the 3000-mile range turbojet missile boosted to a Mach number of 0.4 at sea level, a rocket of about 53,000 pounds thrust with a 10-second duration and a weight of 2500 pounds is required, again using the gasoline-oxygen propellant combination.

For the 6000-mile range ballistic rocket, the higher energy propellant combination of ammonia and fluorine was assumed. The specific impulse of this combination varied from about 275 at sea level to 370 at extremely high altitude. For this 2-stage, 3-unit missile arrangement a total thrust of between 200,000 and 300,000 pounds and a burning duration of 120 seconds for the first stage and 60 more seconds for the second stage are involved. The three units have a total weight of between 120,000 and 200,000 pounds.

How realistic are such rocket engines, what if any are the chances of improving their performance, and what are the principal problems to be solved? Tentative answers to these questions are given in the next paper, which summarizes the current status of the rocket propulsion technique.

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8. STATUS OF THE LIQUID PROPELLANT ROCKET ENGINE

By Gerald Morrell and Vearl N. Huff

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8. - STATUS OF THE LIQUID PROPELLANT ROCKET ENGINE

By Gerald Morrell and Vearl N. Huff

Power-plant requirements for long-range guided missiles and rocket boosters for air-breathing engines have been outlined in the first paper of this series. For the case of the rocket power plant, as high a specific impulse as possible is required, for in this way low missile weights can be obtained. It will be recalled, for example, that the use of a high-specific-impulse propellant system, ammonia and fluorine, results in a missile weight approximately 50 percent of that which can be attained when a low-specific-impulse propellant system, gasoline and oxygen, is utilized. The major questions that must be answered when such a power plant is designed are: (1) What propellant system shall be selected for this engine; (2) what is the best procedure for injecting and burning these propellants; and (3) what engine design will be best for this application? (It must be remembered that the operational life is to be quite short, say of the order of 5 min including preliminary checking.)

In a review of the research and development effort to date in several phases of rocket power-plant design, such as propellant selection, injector design, combustion, cooling, and materials of construction, the areas can be assessed in which additional research effort, beyond that now being expended, is needed to answer these questions within the framework of requirements set up for long-range guided-missile power plants.

In any particular case, final selection of a propellant combination is the result of a compromise between performance, logistics, allowable development time and cost, and missile cost. To obtain the desired high specific impulse, the designer can specify a high-energy propellant system, that is, one having a high ratio of chamber temperature to molecular weight T_c/M, or he can specify a high chamber pressure (figs. 1 and 2, respectively). In figure 1 the solid curve represents a plot of a simplified specific impulse equation (reference 1) for $\gamma = 1.2$, a value which is common for rocket propellant combustion gases. More rigorously calculated values for several propellant systems are spotted on figure 1. As shown in figure 2, the effect of increasing chamber pressure is to increase specific impulse (reference 2). Even a low-specific-impulse propellant system, gasoline and nitric acid, is capable of theoretical specific impulse greater than 250 pounds thrust per pound per second. At the present time experience has been accumulated with chamber pressures as high as approximately 1200 pounds per square inch. In general, performance tends to increase, as predicted by these theoretical curves. A missile can be designed to do the job with almost

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any propellant combination. As was pointed out in the first paper, however, the higher the specific impulse the lower will be the weight of the missile and for this reason, of course, it is desirable to obtain as high a specific impulse as possible from a power plant. For the case of the rocket booster for air-breathing engines, specific impulse is not so important a criterion since in this case the operational range is quite short.

Major engine development effort to date has been in the lower-specific-impulse regions, that is, in regions of specific impulse less than 250 pounds thrust per pound per second. Experience in the higher-specific-impulse regions can hardly be considered more than experimental, and it is in these regions that more intensive research is needed to meet the requirements for long-range guided-missile power plants.

Because of the greater experience that has been accumulated with lower-specific-impulse propellant systems and for reasons of availability and logistics, certain of these propellant combinations can still be considered for use in long-range guided missiles as well as in booster engines. At a recent meeting of the NACA Special Subcommittee on Rockets it was recommended that the lower-specific-impulse systems which should receive attention for these applications are alcohol-oxygen, ammonia-oxygen, and gasoline-oxygen. While the first two systems are being developed more or less intensively, development of the gasoline-oxygen system has lagged. This system is quite attractive from a logistic standpoint and deserves a great deal more research effort.

The high-specific-impulse propellant systems are of greatest promise for application in long-range missile propulsion; recommendations for propellant systems which require development include hydrogen-oxygen, hydrogen-fluorine, and ammonia-fluorine. Experience with these systems is still in the early experimental stages, but the performance obtained to date is encouraging. With the hydrogen-oxygen system, other laboratories (JPL, Aerojet, and Ohio State) have obtained 96 to 97 percent of the theoretical specific impulse calculated for equilibrium expansion; that is, maintenance of chemical equilibrium is assumed during the expansion process. Experiments with the hydrogen-fluorine system in a 100-pound-thrust engine at JPL have yielded equally good performance.

Performance data obtained with ammonia-fluorine at this laboratory at the 100-pound thrust level are shown in figure 3. Peak performance with the 1-to-1 impinging-jet injector is 85 percent of theoretical, but the 2-to-1 impinging-jet injector promises to give much better results. Performance data obtained at this laboratory for the hydrazine-ammonia-fluorine system in the same type of engine are shown in figure 4. Peak performance here is about 90 percent of theoretical. The reason for investigating a mixture of hydrazine and ammonia is that in this way

the higher performance of hydrazine is combined with the low-freezing-point characteristics of ammonia.

The encouraging experimental results obtained to date at low thrust levels indicate that these high-specific-impulse propellant systems are worthy of more intensive research effort. The advisability or the desirability of handling such a reactive material as fluorine may be seriously questioned. Experience at this laboratory and elsewhere, however, has shown liquid fluorine does not represent a serious handling problem. At this laboratory, gaseous fluorine is condensed in propellant tanks suspended in a liquid nitrogen bath and helium pressure is used to transfer the liquid fluorine from peopellant tank to thrust cylinder. No serious corrosion difficulties have been encountered with monel, nickel, or even copper. On the combustion side, the problems involved in handling propellant systems which result in combustion-chamber temperatures of the order of 7000° F are not so great as might be expected. Provided adequate cooling is available, and this does not appear to be too serious a problem at the moment, these hot combustion gases can be handled even by plain carbon steel. On the basis of this experience, it is felt that intensive research with the high-energy propellant system need not be deferred because of anticipated handling difficulties.

Once a propellant system is selected, the next step is usually the selection of an injector design that will give the best performance possible with this particular combination. It is perhaps significant that roughly 85 to 90 percent of the engines developed to date employ the impinging-jet method of injection. This method is illustrated in figure 5, where it is compared with the shower-head method of injection. Since selection of an impinging-jet injector represents the end result of a more or less comprehensive development program in each case, it is probably safe to say that at the present time the impinging-jet method of injection offers the best compromise between high performance, low heat-transfer rates, and stable operation. Indeed, experiments at another laboratory (North American Aviation, Inc.) with alcohol and oxygen indicate that this method of injection gives better and more stable performance than either the shower-head or spray-nozzle methods of injection. Although experience with high-energy propellants is still meager, it appears that even for such a reactive system as ammonia-hydrazine-fluorine the type of injector employed has a marked influence on performance (fig. 6). Throughout the range of fuel-oxidant ratios investigated, the impinging-jet injector gave higher performance than the shower-head injector. These conclusions should not be construed as applying to all propellant systems, nor should the possibility be ignored that still better injection processes may eventually be developed.

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For a given type of injector, certain design parameters influence the performance which can be obtained in an engine. Reaction Motors, Inc., for example, has studied the injector design parameters which influence performance for the special case of a 1-to-1 impinging-jet alcohol-oxygen injector at the 200-pound thrust level (reference 3). An example of the data obtained is shown in figure 7. It will be noted that the shorter the stream length the better the performance, stream length being defined here as the distance from the injector face to the point of impingement. Up to included angles of 60°, between-the-jets performance increased with increasing impingement angle. Above 60°, however, impingement angle appears to have little effect. At this point a word of caution is required. As stream length is decreased, erosion and burning of the injector face may occur for some propellants, so that the optimum stream length may not be approached for practical reasons. Injection pressure drop and jet misalignment were among other parameters found to affect performance. Increasing injection pressure drop improved performance, while jet misalignment decreased performance. In the former case, more intensive mixing is probably obtained, while in the latter case the configuration is approaching that of a shower-head injector, which gave poorer performance with this propellant combination. Because the effects of scale-up are not completely known at this time, data of this kind should be extrapolated with caution to higher thrust levels. Studies of this kind, however, over a range of thrusts would be required in the intensive development of the gasoline-oxygen system and certainly for the high-energy propellant systems mentioned previously.

2546-4

Whereas the injector acts as a mixture preparation device, the combustion chamber provides the necessary volume for completion of the combustion reaction. For most of the engines developed to date, combustion efficiencies are in the range of 85 to 95 percent, which indicates that rather high combustion efficiencies are possible in the rocket chamber. Increased research effort on the high-energy propellant systems is expected to result in equally high combustion efficiencies. It should be remembered that for the rocket engine, combustion efficiency is independent of operational altitude; one of the problems inherent in air-breathing engines is thereby eliminated.

There are still some unsolved problems in the combustion field, however. At some stages of development, many rocket engines exhibit a phenomenon known as vibratory or unstable combustion. The pressure fluctuations accompanying resonance burning may be great enough to over-stress the thrust cylinder, or, more commonly, the heat rejection rate may increase beyond the capacity of the cooling system. Because of these destructive end effects, considerable development time is often consumed in eliminating unstable combustion when it is encountered even though performance may not be adversely affected.

[redacted]

2546-M

Analyses of low-frequency combustion vibrations, commonly called chugging, are usually based on the following concept (references 4 and 5): If it is assumed that the chamber pressure fluctuates, the propellant flow into the thrust chamber will also fluctuate since flow depends on the pressure differential between tank and thrust cylinder. Differential equations relating propellant flow and chamber pressure are set up, and the coupling force is assumed to be a combustion time delay, that is, the time interval between injection and burning of the propellants. Under proper conditions, this combustion time delay acts as a feedback device which can throw the system into resonance. This analysis is in fairly good agreement with the available data and suggests methods for eliminating low-frequency combustion vibrations. Some of these methods are as follows:

1. Combustion time delay may be decreased by changing injector design or by use of additives in the fuel.

2. The damping may be increased by increasing the resistance of the system, that is, by increasing the ratio of feed system pressure drop to chamber pressure.

3. The propellant feed line inertia may be changed by increasing the length of the line or the flow velocity in the line.

The problem presented by high-frequency combustion vibrations, commonly called screaming, are presently considered more serious than those associated with the lower-frequency vibrations since present knowledge concerning high-frequency combustion vibrations is in a less satisfactory state. Then too, this type of combustion vibration appears to be occurring more frequently as major power plants operating at higher chamber pressures are developed. Although at least one analysis has given fair agreement between predicted and measured frequencies in an alcohol-oxygen engine, the problem is still not completely defined, nor are methods for eliminating the phenomenon apparent from these analyses.

In order to define the problem more clearly, data were collected from a number of rocket engine manufacturers on case histories of high-frequency unstable combustion. Analysis of these data indicates that high-frequency resonance burning can occur with almost all engine designs and pumping systems. Surprisingly enough, a major portion of the data agrees with the concept of axial-thrust cylinder resonance as shown in figures 8 and 9. Figure 8 represents the concept considered whereas figure 9 shows how well the data fit the curves plotted on the assumption that frequency is 0.8 of the theoretical frequency for constant chamber temperature. Several causes for the initiation of high-frequency resonance burning have been advanced. For example, experiments at this laboratory with impinging jets (reference 6) have

shown that the resultant sheet of liquid breaks up into waves of droplets with a frequency of 1000 to 4000 cycles per second and it is possible that this behavior contributes to initiation and maintenance of resonance burning. On the other hand, if a major portion of the combustion reaction is assumed to take place at a position corresponding to a pressure antinode, the energy release, if sufficiently high, could act as a feedback system supporting resonance burning at some harmonic of the fundamental frequency of the tube. High-speed photographic studies made at this laboratory and elsewhere (references 7 and 8) in transparent walled rocket engines tend to support this latter view of the source of high-frequency combustion vibrations. It would appear then that spreading the combustion zone would eliminate this form of instability. Research effort on this general problem is being considerably accelerated at this laboratory and throughout the industry, and promises to provide at least a method for eliminating this form of unstable combustion when it is encountered.

2546-H

In addition to injector and combustion problems, the designer is also faced with the problem of providing enough cooling capacity to accommodate the high heat-transfer rates characteristic of rocket engine operation, that is, 2 to 10 Btu per square inch per second.

The problem can conveniently be divided into three phases: (1) prediction of heat-transfer rates for a given propellant system and thrust cylinder geometry, (2) design of the cooling system for stable operation, and (3) selection of materials to withstand the high combustion temperatures.

The state of knowledge with regard to calculation of heat-transfer rates in rocket engines has recently been summarized (reference 9). For the convective portion of the heat transfer, the customary empirical equations can be used with an accuracy of 20 to 25 percent. The discrepancies are probably due to the fact that in the rocket engine, flow conditions are not the same as those for which these equations have been developed. In addition, gas properties are not well known at the high temperatures involved. The radiation component is even more difficult to calculate accurately, since extreme extrapolation of gas emissivities is required, and for propellant systems based on fluorine even preliminary estimates of emissivity are not available. As a result of inaccuracies in present methods for calculating heat-transfer rates, considerable redesign of the cooling system may be required as development proceeds. Research programs now under way should supply many of the missing data on gas properties as well as the high-pressure emissivity values for gases.

With regard to cooling system design, the technique of regenerative cooling, that is circulation of one of the propellants in a jacket around the thrust cylinder prior to injection, has proved quite satisfactory for moderate-size engines in the lower-specific-impulse engines.

In high-chamber-pressure engines of the sizes required for long-range guided-missile propulsion, supplementary cooling is often required and this usually takes the form of internal film cooling, the principle of which is shown in figure 10. It will be noted that coolant is injected at discrete stations on the inner wall of the combustion chamber. Evaporation of the coolant tends to keep the wall temperature at or near the saturation temperature of the coolant. In order to maintain uniform wall temperature, multiple injection may be necessary.

Experiments at this laboratory in straight ducts with inert liquids as coolants (reference 10) have established some of the variables which influence coolant effectiveness as shown in figure 11. Liquid-cooled length has been found to be a function of gas and coolant properties, gas flow conditions, and coolant flow. Coolant effectiveness decreases with increased coolant flow, presumably because of increased liquid film turbulence (reference 11). In another series of experiments (reference 12) the chamber of a 1000-pound-thrust ammonia-oxygen engine was film-cooled with ammonia with no regenerative cooling and with no loss in performance; structural weight savings may thus be possible through use of this method of cooling. Internal film cooling is not suitable for complete cooling of the nozzle, however, since experiments with this same type of engine have resulted in a 23-percent decrease in performance. Continuing research effort at this laboratory and elsewhere appears sufficient to supply the basic information needed for missile power-plant cooling-system design.

Assuming that adequate cooling is provided, it would appear that highly alloyed metals are not needed for the thrust cylinder in view of the short operational life required. For example, plain carbon steel has sufficient corrosion resistance to be usable up to 1500° F (reference 13).

As an alternative to providing supplementary cooling for large missile engines, the use of ceramic liners or coatings to reduce heat-transfer rates to the regenerative cooling jacket appears attractive. The principle involved is shown in figure 12. In this figure the thickness of the ceramic lining is not meant to be indicative of actual practice, since the optimum thickness would have to be determined by research. As a matter of fact, materials can be added to the fuel which during combustion will be deposited as a thin coating on the chamber walls; this type of coating can be quite effective in reducing heat-rejection rates. It should be remembered that the attractiveness of these liners and coatings arises out of the short operational life requirements and they cannot be depended on to provide a universal solution to the problem of high heat-transfer rates. The effectiveness of ceramic liners is illustrated by a set of experiments (reference 9) wherein heat-transfer rate in the combustion chamber was

2546 - M

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considerably reduced with a zirconia liner which had an operational life up to 4 minutes. It would appear that expansion of the research effort for fabrication and testing of ceramic liners and coatings for large cooled engines could result in a simplified cooling problem for the rocket-engine designer.

Past research and development effort has been extensive rather than intensive; some experience has been gained with almost all propellant combinations and all phases of engine design. Intensive development has been confined to the alcohol-oxygen and gasoline-acid systems, the latter being primarily of interest for assisted-take-off use and air-to-air missile use. In order to meet the requirements for aircraft boosters and long-range guided-missile power plants, additional research effort is needed in the following areas: (1) Intensive development of the gasoline-oxygen system because of its advantageous logistic position, (2) intensive development of higher-energy propellant systems because of potential savings in structural and gross weights of long-range missiles, (3) a solution of the problem of high-frequency combustion vibrations in order to facilitate rocket engine development and reduce development cost, and (4) application testing of ceramic liners or coatings for reducing heat-transfer rates in high-pressure, large-thrust engines.

While not discussed for lack of space, certain other problems also require additional attention. Among these are problems involving propellant pumping, thrust cylinder fabrication techniques, and power-plant controls.

In conclusion, it is probably safe to say that none of the problems discussed are of the type which could block development of long-range missile power plants; rather they are the type that would increase development time and manpower requirements and hence increase development costs.

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SPECIFIC IMPULSE AS FUNCTION OF T_C/M

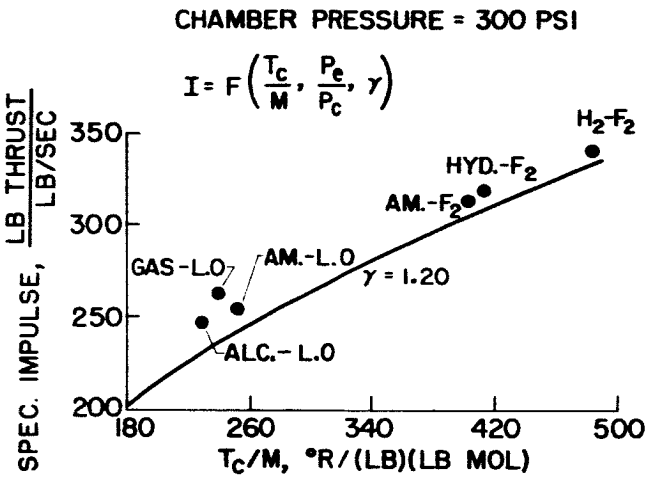

Figure 1

SPECIFIC IMPULSE AS FUNCTION OF CHAMBER PRESSURE FOR EXPANSION TO SEA-LEVEL PRESSURE

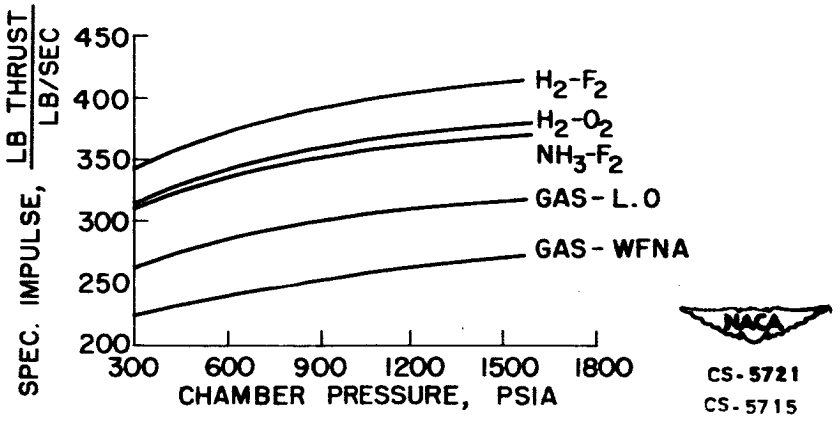

Figure 2

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PERFORMANCE OF AMMONIA-FLUORINE SYSTEM IN 50 L* ENGINE

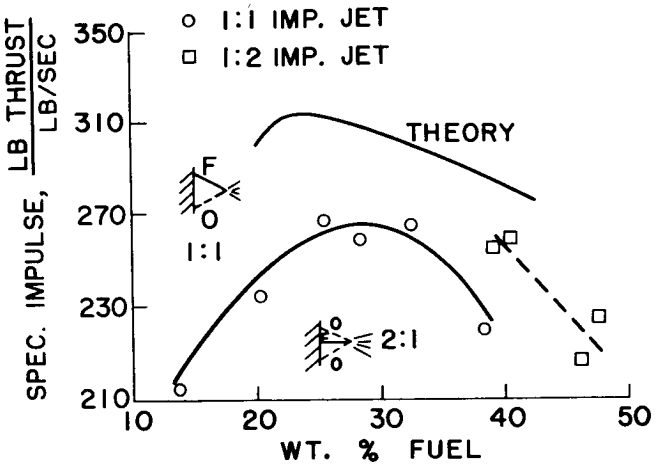

Figure 3

PERFORMANCE OF 36% AMMONIA-64% HYDRAZINE FLUORINE SYSTEM

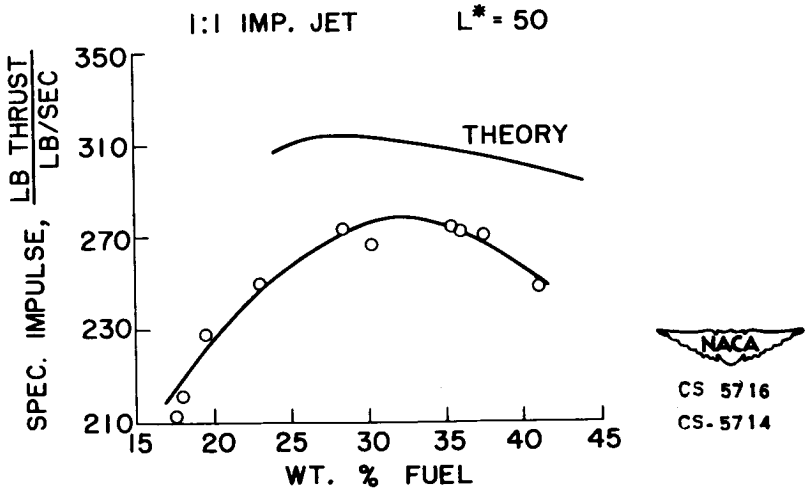

Figure 4

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COMPARISON OF SHOWER-HEAD AND IMPINGING-JET INJECTOR PRINCIPLES

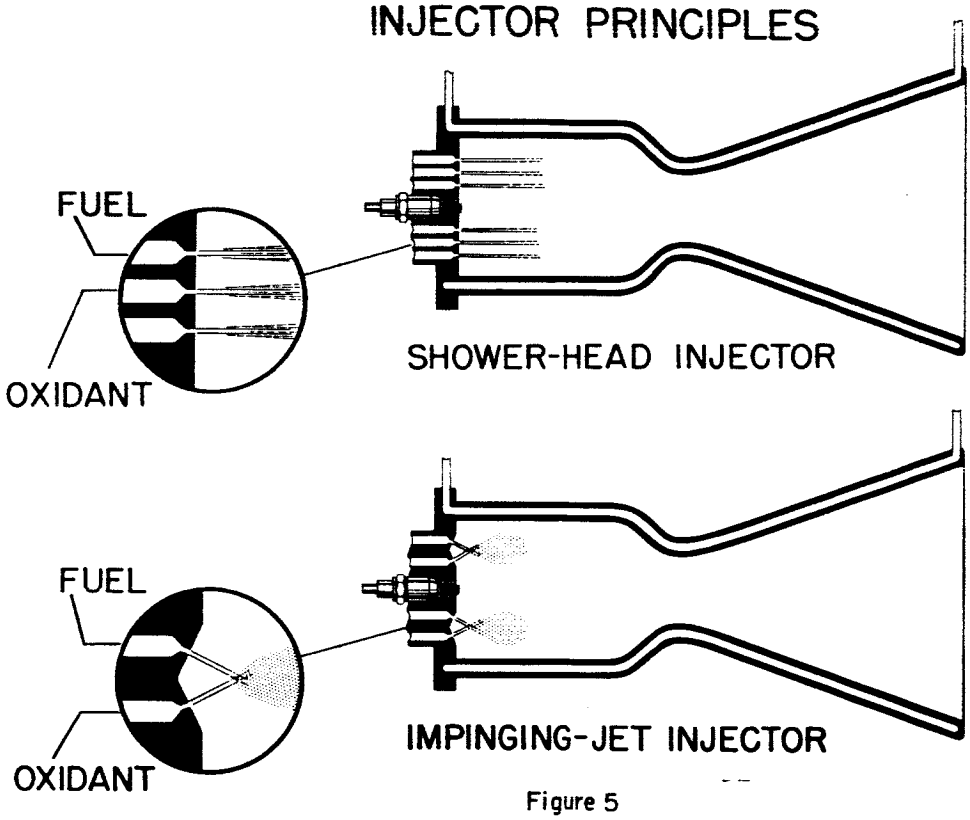

Figure 5

PERFORMANCE OF AMMONIA-HYDRAZINE-FLUORINE SYSTEM AS AFFECTED BY INJECTOR DESIGN

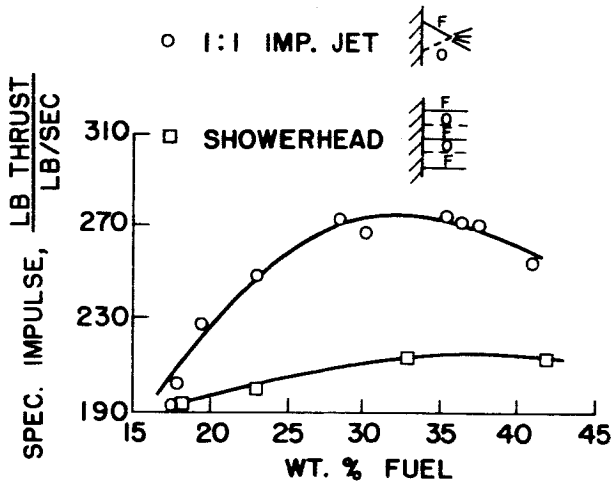

Figure 6

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PERFORMANCE OF ALCOHOL-LIQUID OXYGEN SYSTEM AS AFFECTED BY JET STREAM LENGTH (RMI DATA)

O/F = 1.4 n = 18 INJ. ΔP = 50 PSI
P_c = 300 PSI THRUST = 200 LB

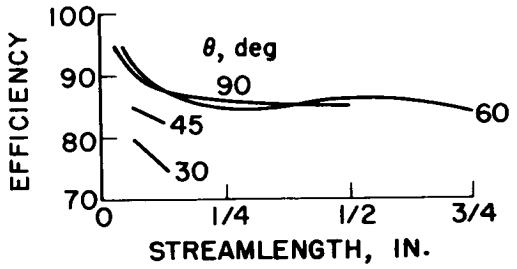
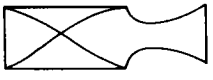


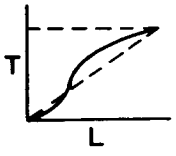
Figure 7

ANALYSIS OF AXIAL COMBUSTION-CHAMBER RESONANCE



ASSUMING LOW AMPLITUDE WAVE AND CONSTANT TEMPERATURE:

$$f_{th} = \frac{na_c}{2L} = \frac{n\sqrt{\gamma g R T_c}}{2L}$$



FOR LINEAR TEMPERATURE VARIATION: $f = 0.7 f_{th}$

ASSUMPTION OF $f = 0.8 f_{th}$ FITS DATA QUITE CLOSELY

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Figure 8

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INFLUENCE OF CHAMBER LENGTH ON RESONANT COMBUSTION FREQUENCY

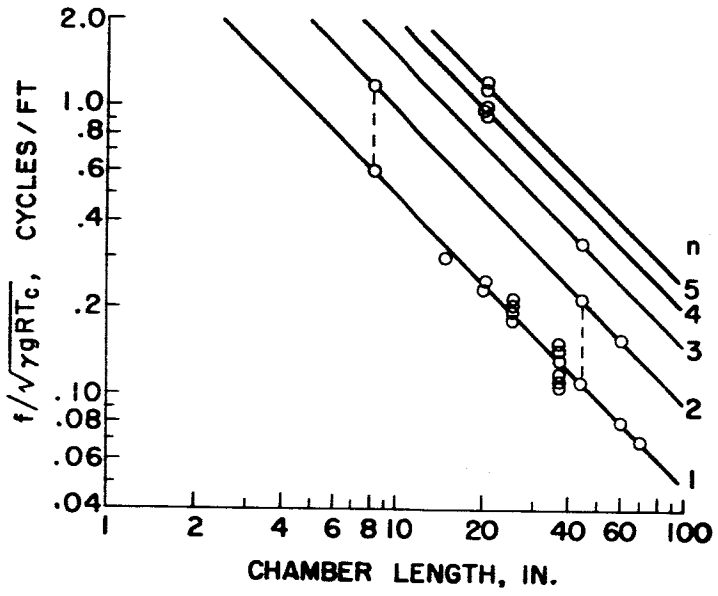
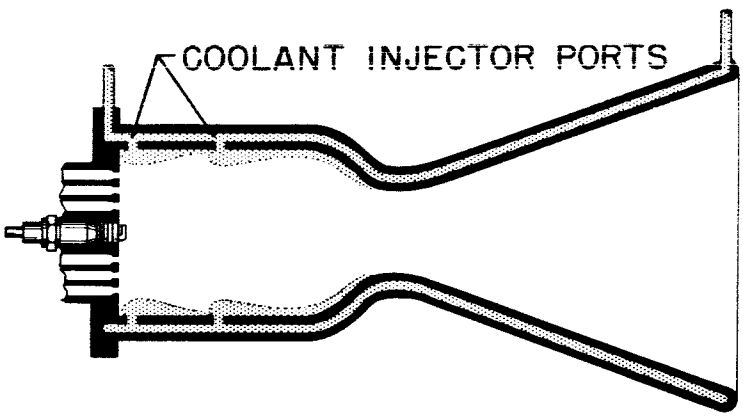


Figure 9

ILLUSTRATION OF INTERNAL FILM COOLING PRINCIPLE



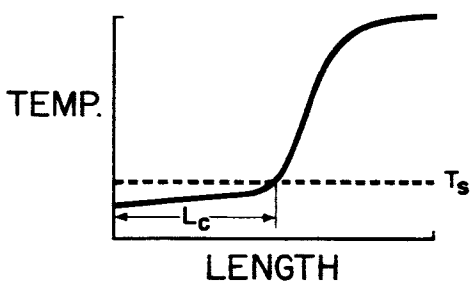


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Figure 10

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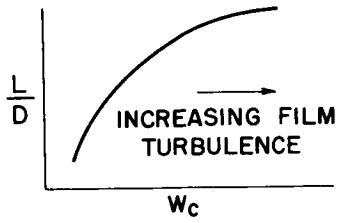
FACTORS THAT INFLUENCE FILM-COOLING EFFECTIVENESS



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CS-5778

$\frac{L}{D}$ IS FUNCTION OF GAS AND COOLANT PROPERTIES, AND GAS FLOW CONDITIONS, AS WELL AS COOLANT FLOW

Figure 11

APPLICATION OF CERAMIC LINER TO REDUCE HEAT TRANSFER RATE

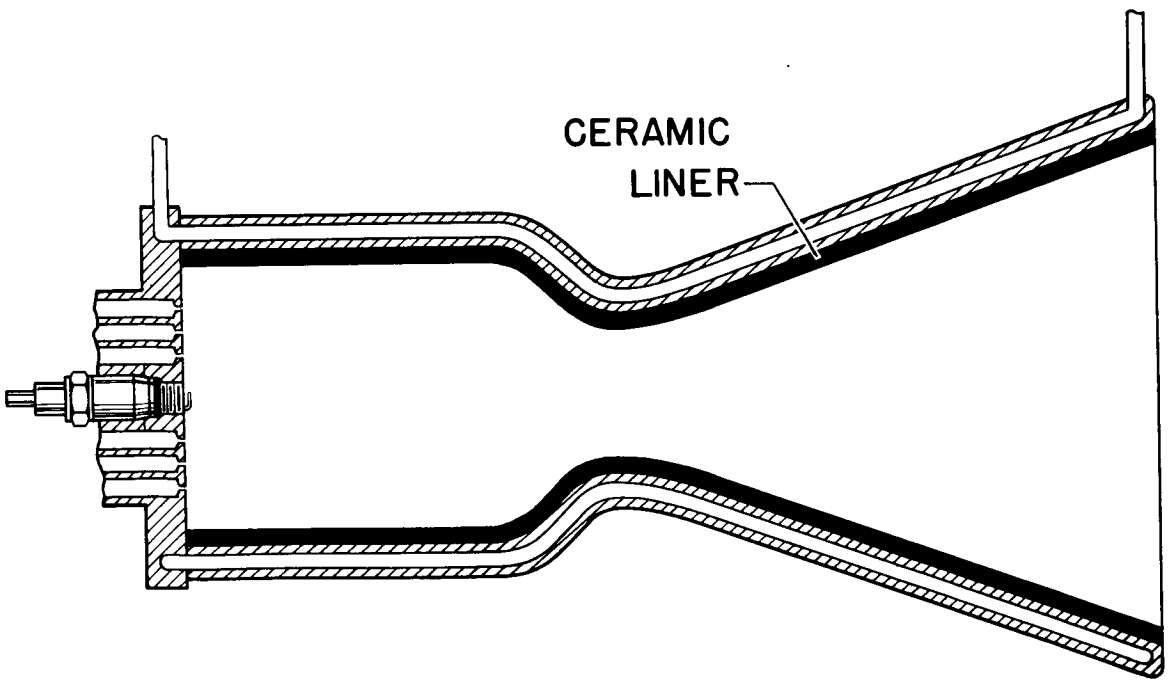

Figure 12

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9. INTRODUCTION AND COMMENTS IIa

By DeMarquis D. Wyatt

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9. - INTRODUCTION AND COMMENTS IIa

By DeMarquis D. Wyatt

In the morning session, the analytical requirements for optimum powerplant performance of supersonic missiles were analyzed. In this analysis, ram-jet and turbojet powered missiles were considered independently. There are, however, numerous items in the powerplant installation picture which have a common significance and effect on either engine type, and some of these items will be discussed in the afternoon session.

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One of the common factors influencing missile performance with either engine type is the location of the engine, or engines, in the missile. Unfortunately, when missile performance is estimated at the design stage, it is difficult to predict all the interference or interaction effects between missile components. Consequently it is difficult to evaluate the aero- and thermodynamic effects of alternate engine locations with the same degree of assurance that can be used to evaluate such factors as the relative engine accessibilities and structural weights associated with each position.

An experimental approach to the evaluation of the relative merits of various engine locations is likewise difficult because of the many independent variables in a missile design which might influence the answers. However, in an effort to obtain some of the necessary data, an experimental program is now underway in the Lewis 8- by 6-foot supersonic tunnel. For this program, a long-range ram-jet missile flight plan was assumed which called for a Breuget flight path at a Mach number of 2.0 at an initial altitude of 50,000 feet with an initial missile gross weight, exclusive of boost, of 50,000 pounds. Several models having basically different engine locations were then designed to meet the assumed requirements. These models were not intended to represent particular concepts for detailed missile designs, but rather to include the composite problems that might arise from each engine location.

Two of these configurations have now been investigated. The first model (fig. 1) represents configurations having engines pylon-mounted from the fuselage. Preliminary investigations of the engines and body without the wings or control surfaces showed that favorable drag interference effects occurred with the engines located close in and rearward on the fuselage (reference 1). This arrangement was therefore adopted for the complete model.

The second configuration investigated (fig. 2) represents an engine installed in the fuselage with an underslung scoop inlet located midway

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on the fuselage. The accumulated boundary layer was removed just ahead of the inlet by a ram scoop depressed into the fuselage. The inducted boundary-layer air was discharged from flapped gills located under the wing at the root. No experimental effort was made to optimize this design.

The drags for these two models at a Mach number of 2.0 are compared in figure 3 as a function of angle of attack. The model drags were within 4 percent of each other through the angle of attack range up to about 5°, with the engine-in-fuselage configuration having the lower drag. The cruise angle of attack is indicated for each model, together with the cruise lift-drag ratio. Besides having a slightly lower drag, the engine-in-fuselage configuration also had a slightly higher lift-curve slope, probably because of greater body lift on the cylindrical afterbody, and therefore a higher cruise lift-drag ratio. The differences, however, are not large.

2546-I

These data, although fragmentary, indicate that it may be possible to have markedly different engine installations for identical missile missions without large differences in configuration drags and therefore in thrust requirements. Other arrangements are now being investigated in an effort to shed further light on this question.

If there are no major drag differences with different engine installations, other factors will become primary in determining an optimum configuration. With respect to an engine, one of these principal factors will necessarily be the pressure recoveries attainable with the inlet systems. At previous NACA conferences, the status of inlets designed for supersonic flight up to a Mach number of 2 has been considered in some detail, and it was concluded that in general, the inlet picture is well in hand for this range. It has been shown, for example, that scoop inlet pressure recoveries about equal to nose inlet values can be achieved in this Mach number range if adequate boundary-layer-removal techniques are utilized (references 2 and 3). The inlet system for the model shown in figure 2 yielded a maximum pressure recovery of about 84 percent at Mach number 2 in confirmation of the previous findings.

The analysis presented by Mr. Hall in the morning session indicated that optimum long-range missile flight will be achieved at higher Mach numbers than 2. It is therefore of interest to examine the status of diffuser research at these Mach numbers. There has been considerable research at Mach numbers up to about 3.1 as indicated on figure 4, which shows peak pressure recoveries achieved in about 50 tests by different investigations in this country. Superimposed on these data are lines of constant kinetic energy efficiency η_{KE}. This efficiency is defined as the ratio of kinetic energy potentially available after diffusion to the kinetic energy in the free stream and is significant

in estimating the specific fuel consumption potentialities of the ramjet engine. As indicated by the equation on figure 4, the specific fuel consumption can be approximately expressed as a simple function of the kinetic energy efficiency. From this equation it can be shown that small changes in kinetic energy efficiency will result in small changes in specific fuel consumption for otherwise constant operating conditions. Furthermore, although the pressure recoveries are steadily decreasing as the Mach number is increased along a constant kinetic energy efficiency line, the increases in specific fuel consumption from ideal values remain approximately constant for fixed fuel-air ratios and combustion efficiencies. Thus, the fuel penalty at a Mach number of 4.0 with a 0.60 pressure recovery would be only about the same, percentagewise, as the penalty at Mach number 2.0 with 0.87 pressure recovery.

2546-J

The existing data are pretty well defined by a kinetic energy efficiency of 0.95 as an upper limit. The few points above this line are all from isentropic-cone or perforated convergent-divergent diffusers, and from the standpoint of useful application must be qualified by drag and airflow stability considerations, as will be shown. There are practically no published data on diffuser performance above Mach number 3.1. At this laboratory an intensive exploratory program has been conducted at Mach number 3.85, as well as some work near Mach number 3.0. In the next paper, the results of this research are described and correlated with the existing data.

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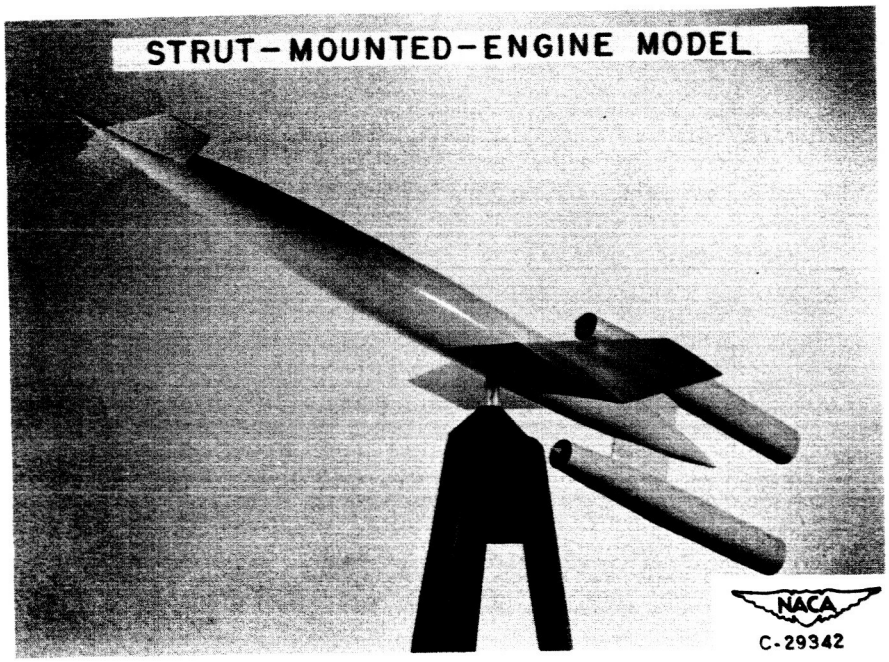

Figure 1

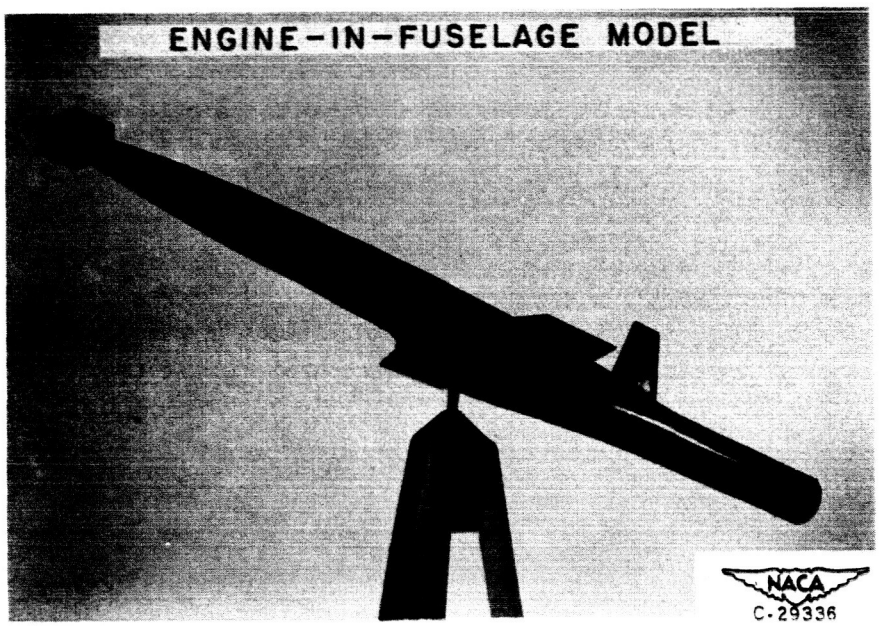

Figure 2

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CONFIGURATION DRAGS
M_O, 2

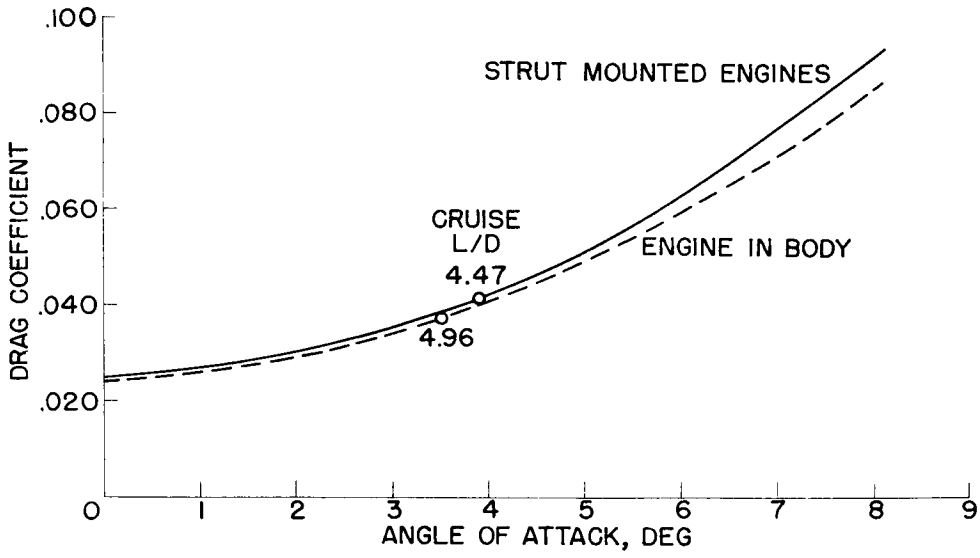

Figure 3

SUMMARY OF DIFFUSER RESEARCH

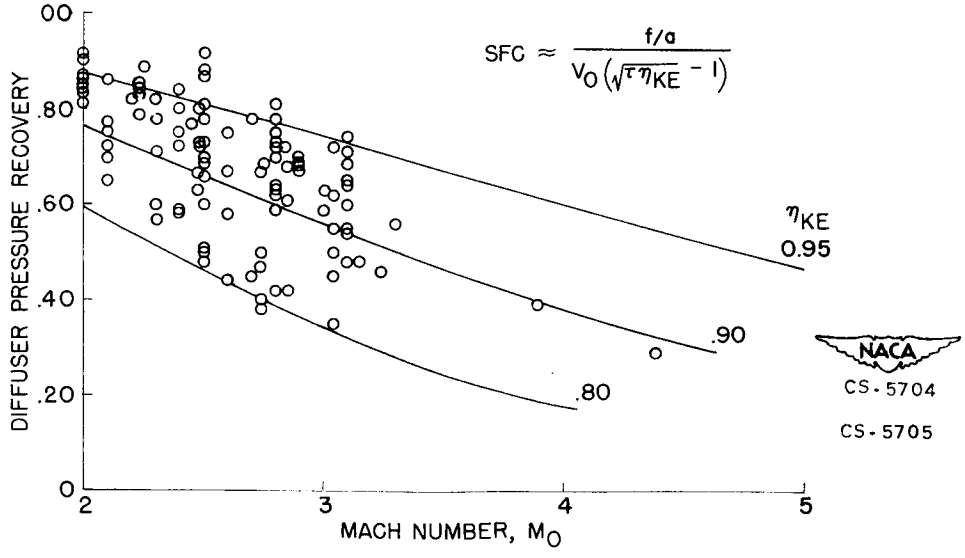

Figure 4

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10. SUPERSONIC INLETS AT HIGH MACH NUMBERS

By Edgar M. Cortright, Jr.

and James F. Connors

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10. - SUPERSONIC INLETS AT HIGH MACH NUMBERS

By Edgar M. Cortright, Jr. and James F. Connors

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Design studies of long-range supersonic missiles indicate the Mach number range of 3 to 4 to be of considerable promise. Accordingly, the NACA is conducting research on the performance of a wide variety of supersonic diffusers in this range. The initial phase of this research is concerned primarily with the characteristics of conventional axially symmetric diffusers operating at design values of flight Mach number. In addition, some investigations of side inlets and two-dimensional "split-wing" inlets have been conducted. The present paper will summarize the results of this preliminary research in the light of missile-inlet requirements.

The supersonic diffuser types on which the majority of this discussion will be based are shown in figure 1. All these diffusers utilize a projecting centerbody to create external compression ahead of the terminal shock wave which is located at or near the diffuser throat. The 1-cone diffuser is so designated because it utilizes a single conical surface to generate a compression wave ahead of the throat. A modification of this inlet, the 1-cone (low-drag cowl) inlet, incorporates a high rate of turning at the throat so as to reduce the cowl lip angle and virtually eliminate the external pressure drag. Since there are no existing criteria as to how rapidly the flow at the entrance may be turned back toward the engine axis, this inlet was designed to represent a limiting case. In order to make the inlet function properly (with normal shock swallowed), it was necessary to eliminate or reduce boundary-layer separation which occurred on the centerbody immediately downstream of the sharp turn. This was accomplished by drilling holes in the centerbody at this point and venting the interior to free-stream static pressure. Boundary-layer air passed into the centerbody through the holes and then discharged into the free stream through hollow centerbody support struts. This action will be discussed more fully at a later point. A third inlet type is the 1-cone inlet with internal contraction designed for additional compression ahead of the terminal shock wave. This inlet is theoretically capable of higher pressure recovery than the 1-cone inlet and will have a lower cowl drag because of decreased cowl lip angles. Another, the 2-cone inlet, utilizes two conical surfaces to generate compression waves ahead of the throat. This diffuser may, of course, have internal contraction and did so in the present investigations.

The so-called isentropic spike diffuser utilizes a continuous compression surface ahead of the throat and theoretically yields the highest total pressure recovery of any of this family of diffusers. The spike tip may be long and slender, as indicated, or may be a cone of moderate angle. The compression surface generally produces large amounts

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of turning and the resultant cowl lip angle may require a detached shock wave and high pressure drag. Use of some internal contraction may alleviate this problem by reducing the lip angle. An interesting approach to the problem of high cowl drags in the case of inlets with large external compression has been proposed by members of the staff of the John Hopkins University (reference 1). This streamlined or shielded isentropic inlet is fitted with an annular extension of low drag profile supported ahead of the cowl proper. The shield essentially converts the inlet into a supersonic diffuser with large internal contraction. Swallowing of the starting shock wave is permitted by spilling flow through the annular opening. Once supersonic flow is established up to the throat, careful contouring of the cowl might provide a pressure balance across the dead air region, permitting little or no spillage of captured air. This has been achieved with varying success at lower Mach numbers. At a Mach number of 3.85, however, the inlet has not been made to operate without spilling large quantities of air; this inlet is thus unsatisfactory in its present state of development.

The adverse pressure gradients imposed upon the boundary layer generated by the centerbody are very great at the Mach numbers under consideration, particularly in the case of the high compression inlets. In an attempt to prevent extensive boundary-layer separation on the forebody and in the throat, porous centerbodies, fabricated of sintered bronze, have been utilized to remove the boundary layer continuously from the spike tip to a station slightly downstream of the diffuser throat. The air is handled in the same way as with the 1-cone inlet with low-drag cowl already discussed.

Combining recent data obtained at Mach numbers of 3.05 and 3.85 with data existing in the literature yields general curves of the variation of total pressure recovery with flight Mach number for the various diffuser types. These variations are shown in figure 2 where the curves generally represent a high mean. Data points are indicated where new data establish the curves. With the exception of the 1-cone type, no distinction is made between inlets with or without internal contraction. Above a Mach number of 1.9 the total pressure recovery of all the diffusers begins to drop off quite rapidly and, in addition, the spread between the curves increases greatly. The pressure recovery increases as the compression is made more nearly isentropic but not quite so much as would be expected from simple shock theory, which, at a Mach number of 3.85, would predict a 36-percent recovery for the 1-cone inlet and a 93-percent recovery for the isentropic spike diffuser. The shielded isentropic spike yielded a reasonably high pressure recovery but, because of its unsatisfactory mass flow characteristics, this inlet will not be discussed further herein.

As a matter of academic interest, the highest recovery at $M_0 = 3.85$ was obtained with a porous isentropic spike. Although schlieren photographs indicated little if any boundary-layer separation in the presence

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of the existing strong adverse pressure gradients, no large increases in pressure recovery have been achieved to date. Future research will include porous centerbodies and cowlings in the internal flow passages.

One important negative result at $M_0 = 3.85$, which is not illustrated by this figure, is the lack of success with inlets with variable internal contraction ratio. With both two and three-dimensional diffusers any attempt, following starting of the inlet, to exceed the limiting starting contraction as set forth by Kantrowitz and Donaldson (reference 2) was met with expulsion of the normal shock wave except in some cases with extensive boundary-layer control. Even in these cases, however, no improvements in pressure recovery were obtained.

Up to this point only peak pressure recoveries have been considered. There are practical difficulties associated with operation at peak pressure recovery. In addition, if the peak recovery occurs at a point of less than maximum mass flow it is generally not desirable to operate at this point. Accordingly, it is of interest to examine the characteristics of various supersonic diffusers, which are illustrated in figure 3 for a flight Mach number of 3.05. Total pressure recovery is plotted as a function of mass flow ratio, defined as the ratio of the mass flow through the engine to the mass flow through a free-stream tube area equal to the projected inlet capture area. With supercritical flow, by definition, the diffuser is operating along a line of constant mass flow with the pressure recovery increasing as the normal shock moves closer to the throat. The maximum mass flow ratio varied considerably among the inlets investigated with the high-recovery inlets generally spilling from 8 to 10 percent of the maximum mass flow, m_0. It is not felt that this spillage is inherent except, possibly, in the case of the isentropic inlet without internal contraction, which had external compression to a low enough Mach number to require shock detachment from the cowl lip. However, of the inlets investigated, which were designed by simple shock theory with maximum internal contraction and without boundary-layer consideration, only the 1-cone inlet could be operated at a mass flow ratio of 1. Any attempt to increase the mass flow of the other inlets by moving the cowlings forward resulted in excessive contraction and movement of the normal shock wave ahead of the inlet.

With movement of the normal shock wave ahead of the inlet, all the diffusers began to buzz; that is, the normal shock wave began to pulse in and out of the inlet. The buzz was accompanied by large fluctuations in the diffuser mass flow and pressure recovery. The dotted portions of the curves represent unsteady flow as measured by manometer boards which tend to damp out and average the measurements. The degree to which it is possible to operate close to the peak recovery point but in the unsteady region varies among the various diffusers and, of course, the presence of flame holders and combustion may alter the buzz characteristics. Actually,

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a serious question exists as to whether combustion may be maintained with strong buzz.

Among the techniques to avoid buzz are two simple but somewhat unsatisfactory expedients. The inlet may be operated slightly supercritically at reduced pressure recovery, allowing a margin for movement of the normal shock wave without travel ahead of the cowl lip. In a turbojet, the reduced pressure recovery reduces the mass flow and thrust and increases specific fuel consumption. With a ram jet the reduced recovery may be allowed for in the engine sizing, but there remains an increase in fuel consumption. Another approach is to operate the diffuser with some supersonic spillage, which in most cases at lower supersonic Mach numbers has been demonstrated to provide a stable subcritical region where the normal shock wave may stand and thus reduce the mass flow without initiating the buzz condition. Although this approach permits operation at peak pressure recovery, it requires some additive drag. At low supersonic Mach numbers the additive drag is relatively small but at high Mach numbers it may not be, particularly in the case of the high-recovery diffusers. The ideal inlet with stable subcritical flow, high recovery, and low drag is not at hand. Some asymmetrical configurations have shown stable characteristics and research in this direction is planned.

Diffuser characteristics at a Mach number of 3.85 are shown in figure 4. The same general characteristics exist, although some additional features may be noted in this figure. A relatively small decrease in mean pressure recovery during buzz was encountered with the low-recovery inlets. The severity of the pulsations as observed by schlieren photographs, however, was not reduced. The 1-cone inlet designed for low drag would not operate with the normal shock swallowed as mentioned previously. Application of boundary-layer suction made operation with a swallowed shock possible, with its associated increase in mass flow and pressure recovery. Recent measurements indicate that only 1 or 2 percent of the mass flow through the inlet was removed by the suction. The 2-cone inlet initially yielded a low maximum mass flow. When roughness was applied to the cone tip, both the inlet pressure recovery and mass flow increased appreciably. Adding roughness to the isentropic spike increased the pressure recovery from 0.57 to 0.61 without appreciably altering the mass flow ratio. The mass flow ratios could not be increased by varying the cowl locations, again as a result of a detached shock in the case of the isentropic inlet, and internal contraction in the case of the other inlets.

Schlieren photographs are presented in figure 5 to illustrate the effects of tip roughness. The 2-cone inlet is shown with and without roughness. Without roughness the boundary layer separated and "bridged" the juncture between the two cones. The action of the roughness was presumably to generate a turbulent boundary layer which largely

eliminated the separation. This phenomenon was previously reported by the University of Southern California (reference 3). A somewhat similar picture is observed in the case of the isentropic spike where an apparent separation was eliminated by the roughness.

The 1-cone inlet with low-drag cowl with and without boundary-layer control at the throat is shown in figure 6. With no suction, separation of the boundary layer at the throat presumably results in excessive internal contraction and a detached shock wave ahead of the inlet. With removal of some of the boundary layer at the throat, the flow attaches and the normal shock wave is swallowed.

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In a consideration of the characteristics of supersonic diffusers, a somewhat different phenomenon may be encountered in the case of two-dimensional diffusers. This is illustrated in figure 7 where total pressure recovery is plotted as a function of mass flow ratio for such a diffuser utilizing an isentropic wedge with internal contraction. Supercritical operation appears similar to that of the three-dimensional counterpart. When the inlet goes subcritical, however, the flow separates on either the top or bottom of the wedge at zero angle of attack and on the top at positive angle of attack. This separation is believed to result from a twin-duct instability in the subsonic diffuser and is accompanied by very large discontinuities in pressure recovery and mass flow. In addition, discontinuities in the wing section characteristics appear. In some cases at lower Mach numbers, this condition has been alleviated by cross-ventilating the subsonic diffuser passages.

A large fraction of diffuser tests and all those discussed to this point have been made at relatively low values of Reynolds number. Observation of roughness effects strongly implies the probability of Reynolds number effects on supersonic diffuser performance. In figure 8, pressure recovery is plotted as a function of Reynolds number for the supersonic diffusers designed for $M_0 = 3.05$ and operating in the variable Reynolds number tunnel at $M_0 = 3.13$. The Reynolds number is based on the inlet diameter at the lip of the cowl and the data approach a value equivalent to a 3-foot-diameter inlet at 80,000 feet altitude. In the lower portion of the figure is shown a family of inlets the pressure recoveries of which increase with increasing Reynolds number. The 1-cone diffuser with rapid internal contraction (throat B as contrasted with a more gradual contraction in throat A) exhibited a discontinuity in its variation. In contrast to results at $M_0 = 3.85$, the presence of roughness lowered the pressure recoveries. In the upper portion of the figure, results with an isentropic spike inlet with no internal contraction are shown. With this inlet the pressure recovery decreased with increasing Reynolds number until roughness was added to the spike tip. Retracting the cowl also altered the characteristics. It is thus indicated that the effect of Reynolds number is dependent on design details and that high Reynolds

number may not necessarily be simulated by tip roughness. However, the variations indicated are not sufficiently large to influence a relative comparison of the inlets.

In any comparison of inlets, the drag is an important factor. In figure 9 are shown the drag coefficients based on maximum cross-sectional area of three ram-jet engines designed for the same net thrust minus drag but utilizing different supersonic diffusers. The cowl pressure drags are experimentally determined, whereas the friction and additive drags are estimated. The 1-cone and 2-cone inlets indicate comparable drag coefficients which amount to slightly more than 20 percent of the engine gross thrust at the indicated value of fuel-air ratio. A small amount of additive drag was encountered with the 2-cone inlet. This drag probably could have been eliminated by redesign at the expense of a slight increase in cowl drag. The isentropic spike inlet exhibited a reduced cowl drag due to the presence of large amounts of flow spillage which caused an expansion at the inlet lip and reduced the cowl pressures but which also caused large additive drag. The magnitude of this additive drag has not yet been determined but may be placed between the minimum and maximum values indicated. Drag of this ram jet would then fall between 29 and 36 percent of the gross thrust. The drag characteristics of the 1-cone inlet with low drag cowl are currently being studied. Preliminary results, which show that only 1 to 2 percent of the flow was removed through the boundary-layer control, indicate a total drag coefficient of less than 60 percent of that of the 1-cone inlet.

By use of these data and the corresponding pressure recoveries and mass flow ratios, it is possible to compare the complete ram-jet engines. Such a comparison is made in figure 10 where sketches of three engines designed for the same thrust minus drag but utilizing different inlets are shown. Exit nozzle expansion to free-stream static pressure is assumed. In general, the engines are quite similar. Use of the high-recovery inlets results in engines of smaller diameter compared with the 1-cone inlet. In the case of the 1-cone inlet, the maximum diameter is governed by the size combustion chamber required for an inlet Mach number of 0.16 which has been assumed. With the 2-cone and isentropic inlets the diameter is governed by either the exit area or the rapidity with which the flow is turned at the throat. If excess cross-sectional area is utilized, the combustion chamber may be enlarged and the chamber Mach number may be reduced for these inlets, which facilitates combustion, as pointed out in an earlier paper by Mr. Breitwieser. Of course, the higher-recovery inlets also have higher combustion-chamber pressures, which may be significant at high altitudes. If only the engine thrusts and drags are considered, the 2-cone inlet and the isentropic inlet with minimum additive drag indicated engines of approximately 12 percent lower specific fuel consumptions than the engine utilizing a 1-cone inlet. With maximum additive drag, however, the isentropic spike inlet is comparable to the 1-cone with respect to specific fuel consumption. It should again be

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recalled that these results are for specific inlets and that the 1-cone inlet investigated did not utilize as low a drag cowling as is possible; hence, the advantage in specific fuel consumption of going to higher recovery inlets may not materialize following optimum development of each diffuser type. This possibility is emphasized by the fact that preliminary data for the 1-cone inlet with low-drag cowling indicate a slightly lower specific fuel consumption than any of the aforementioned cases. Hence, it must be concluded that from a specific fuel consumption standpoint at $M_0 = 3.85$, the low-recovery diffusers are competitive with those of high pressure recovery. In this regard it should be recalled that the diffusers assumed in the missile analysis presented in an earlier paper by Mr. Hall approximated the 1-cone (low-drag cowl) inlet in both performance and geometry.

The discussion has thus far been limited to supersonic diffusers at zero angle of attack. For some missiles such as long-range missiles boosted to design point and flying at constant Mach number and lift to drag ratio, this condition is predominant. Other missiles are forced to maneuver and in such cases the inlet performance at angle of attack must be considered more heavily. In figure 11 total pressure recovery and mass flow ratio are plotted as a function of angle of attack for a group of supersonic diffusers at $M_0 = 3.05$ and another group at $M_0 = 3.85$. At $M_0 = 3.05$ the diffuser total pressure recoveries decrease at different rates with increasing angle of attack. The isentropic spike diffusers fall off in pressure recovery more rapidly than the 1-cone and 2-cone diffusers up to angles of attack of 6° to 7°, at which point the characteristics vary widely. The flow on the upper surface of the original isentropic spike with internal contraction separated (dashed portion) with a sudden large decrease in pressure recovery. Retraction of the cowling eliminated this effect. Separation also occurred on the 2-cone inlet. The corresponding mass flow ratios follow similar trends. All but the isentropic spike with no internal contraction maintain mass flow fairly well until angles of from 6 to 7° are reached. It was observed that the separation also produced a discontinuity in mass flow ratio as would be expected. The elimination of this in the case of the isentropic inlet with internal contraction lowered the mass flow at zero angle of attack.

At $M_0 = 3.85$ similar results were obtained except that the performance of the isentropic spike inlet dropped off even more rapidly as compared with the other diffusers. Separation at high angles of attack was again observed for the isentropic inlet and 2-cone inlet with roughness on the cone tip. It is interesting to note that without roughness the mass flow ratio of the 2-cone inlet actually increased with angle of attack from its low value at zero angle of attack.

The effects of these decreases in mass flow and pressure recovery on the thrust of a given engine at angle of attack are illustrated in figure 12 for a Mach number of 3.85. The engine is assumed to operate

at a fixed fuel-air ratio at constant combustion efficiency and the effects of increased additive drag and increased wave drag at angle of attack are not considered. The ratio of thrust to thrust at zero angle of attack is plotted as a function of angle of attack for several diffuser installations. In the left-hand side of the figure the case of constant combustion chamber inlet velocity M_2, such as would result with a fixed outlet throat area and which might be required with a marginally high value at zero angle of attack, is considered. The decrease in pressure recovery forces a decrease in mass flow in excess of the minimum which could be spilled by the inlet. As a result, all the engines experience a 10-percent thrust loss at $\alpha = 5^\circ$ with the exception of that utilizing an isentropic inlet, which loses almost 30 percent of its zero angle of attack thrust. If the outlet area is increased at angle of attack, excess spillage is not forced upon the inlets. The effect is particularly pronounced in the case of the isentropic inlet, where the losses are reduced to 10 percent at $\alpha = 5^\circ$, and the 2-cone without roughness, where a thrust increase is experienced because of the unusual increase in mass flow already discussed.

Schlieren photographs of several of these inlets at angle of attack are shown in figure 13. With the isentropic spike diffuser at $\alpha = 6^\circ$ at peak pressure recovery, the normal shock wave was observed to stand ahead of the inlet lip on the lower surface. At $\alpha = 9^\circ$ this shock moved still farther ahead, while on the upper surface separation occurred with subsonic and possibly reverse flow at the inlet. In the case of the 2-cone diffuser without roughness, the boundary layer washed towards the top of the cones resulting in decreased bridging on the lower surface and slightly increased bridging on the upper surface. The 1-cone inlet indicates a relatively unchanged flow at angle of attack with the exception of some flow spillage and a slight boundary-layer thickening in the upper half. It may be remembered that this diffuser was least affected by angle of attack.

In the introduction to this paper it was pointed out that, in the Mach number range up to $M = 2$, widely different missile configurations may be aerodynamically comparable and that side inlets may perform as well as nose inlets if adequate removal of the initial boundary layer is provided. Investigation of a single side-inlet configuration has recently been completed at a Mach number of 2.93. In figure 14 pressure recovery is plotted as a function of Mach number for the case of a 1-cone inlet. The data at $M = 1.88$ have already been reported in the literature (reference 4). The supersonic diffuser studied at $M = 1.88$ was modified for the higher Mach number. Removal of the initial boundary layer at $M = 1.88$ yielded a pressure recovery comparable to that of a nose inlet. At $M = 2.93$, however, the recovery, which increased from 0.37 to 0.51, did not closely approach that of a nose inlet. It is of interest to point out that with an initial laminar boundary layer the ram scoop could

not be made to operate in a stable manner. The resulting scoop pulsations spilled about half of the boundary layer into the inlet and resulted in only half of the improvement in recovery indicated in the figure.

The effect of boundary-layer removal on the thrust of a turbojet engine installation is shown in figure 15. Thrust minus drag due to boundary-layer removal is plotted as a function of boundary-layer scoop height parameter, h/δ, where h is the height of the scoop and δ is the thickness of the initial boundary layer at a point where the velocity is 0.99 that of the local free-stream value. The ratios of boundary-layer thickness to inlet radius were approximately 0.15. At $M_0 = 1.88$ with no boundary-layer removal the missile engine would deliver only 60 percent of the thrust possible with a nose inlet. Removal of the boundary layer with no associated drag increased this value to 97 percent. The drags associated with removal either through a duct with a sonic exit nozzle or by means of a wedge to deflect the flow around the inlet (assuming no penalty in inlet performance with removal by the deflection technique) are indicated to be relatively small. The indicated case of maximum drag refers to complete loss of boundary-layer momentum. At a Mach number of 2.93 the thrust with no boundary-layer removal is indicated to be only 42 percent of that possible with a nose inlet. Removal of most of the boundary layer increases this value to 83 percent, although the percentage losses associated with removal are apt to be larger than at the lower Mach number. (With some configurations, removing slightly less than the total boundary layer yields the highest pressure recovery.)

The effects of boundary-layer control on the specific fuel consumption of these installations, shown in figure 16, are less pronounced since the effect of decreased pressure recovery in reducing the mass flow through the engine is not reflected. At a Mach number of 1.88 with no drag considerations, the specific fuel consumption may be reduced by boundary-layer removal from a value 13 percent greater than the nose inlet installation to a comparable value. Drags of boundary-layer removal systems are apt to result in fuel consumptions several percent greater than with a nose inlet. At a Mach number of 2.93 specific fuel consumption is indicated to be improved from a value 25 percent higher than a nose inlet installation to a value 5 percent higher. Drags due to boundary-layer removal are more serious at this Mach number, and fuel consumptions at least 10 percent greater than with the nose inlet have been indicated by the single model investigated. With the limited data available, however, it is impossible to generalize on the results; it is probable that the performance of side inlets of the type investigated can be improved.

In summary it has been shown that supersonic diffusers designed for high total pressure recovery fall increasingly short of their design value as the flight Mach number is increased. Relatively high values of pressure recovery in the high Mach number range have been obtained,

however. High-recovery inlets facilitate the attainment of smaller engines, lower combustion-chamber Mach numbers, and higher service ceilings. Their drags appear higher, however, so that from a specific fuel consumption standpoint, the high- and low-recovery inlets appear to be competitive in the light of present knowledge. On the debit side, the higher-recovery inlets have been found to be somewhat more sensitive to angle of attack. Finally, limited investigations indicate the continued necessity of boundary-layer removal ahead of side inlets as flight Mach numbers increase.

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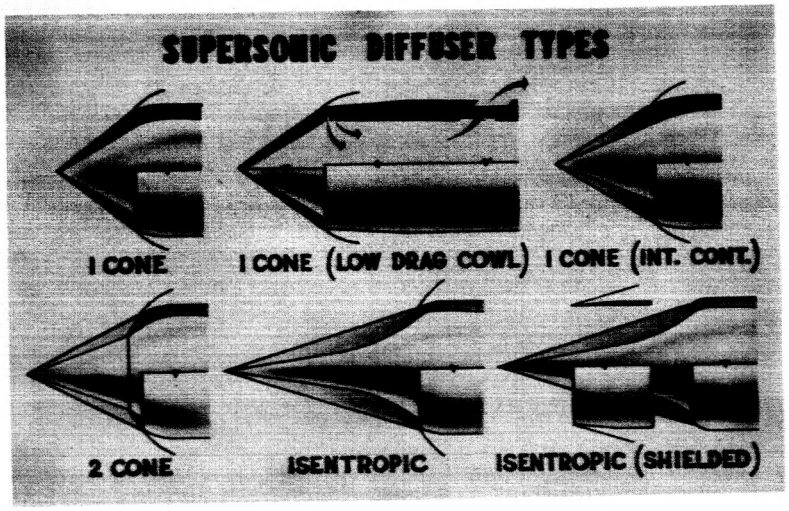

Figure 1

PERFORMANCE OF SUPERSONIC DIFFUSER TYPES

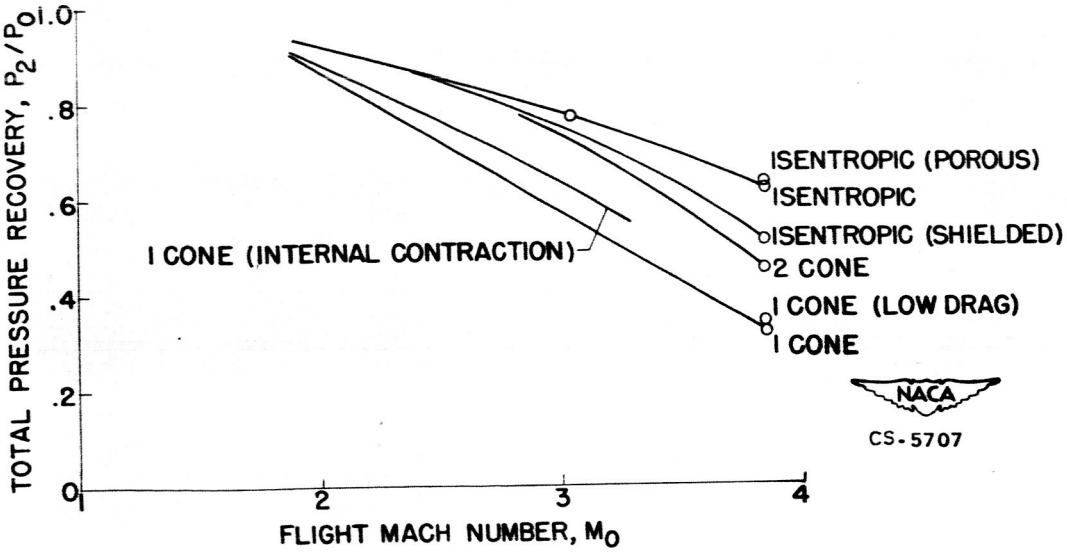

Figure 2

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DIFFUSER CHARACTERISTICS
M_0, 3.05

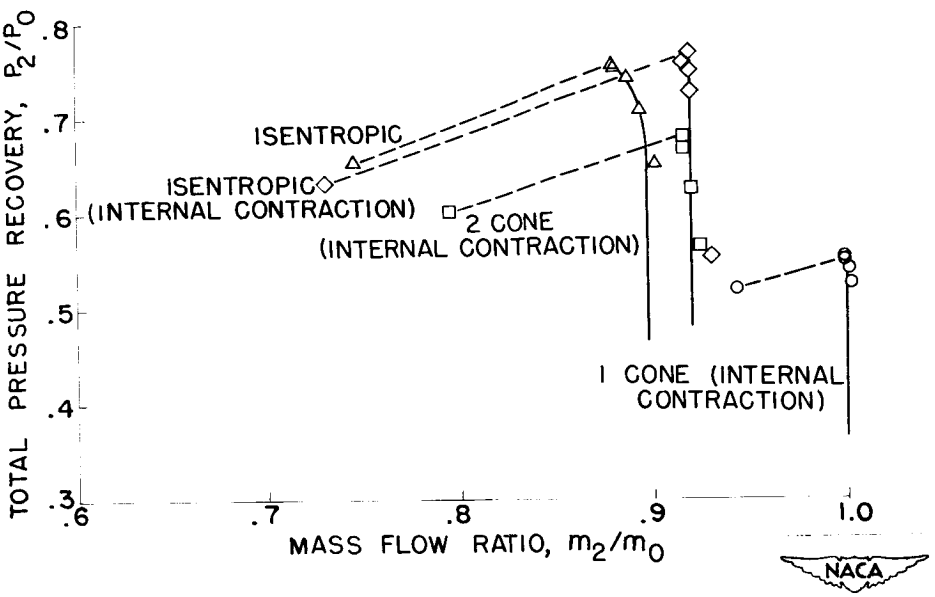

Figure 3

DIFFUSER CHARACTERISTICS
M_0, 3.85

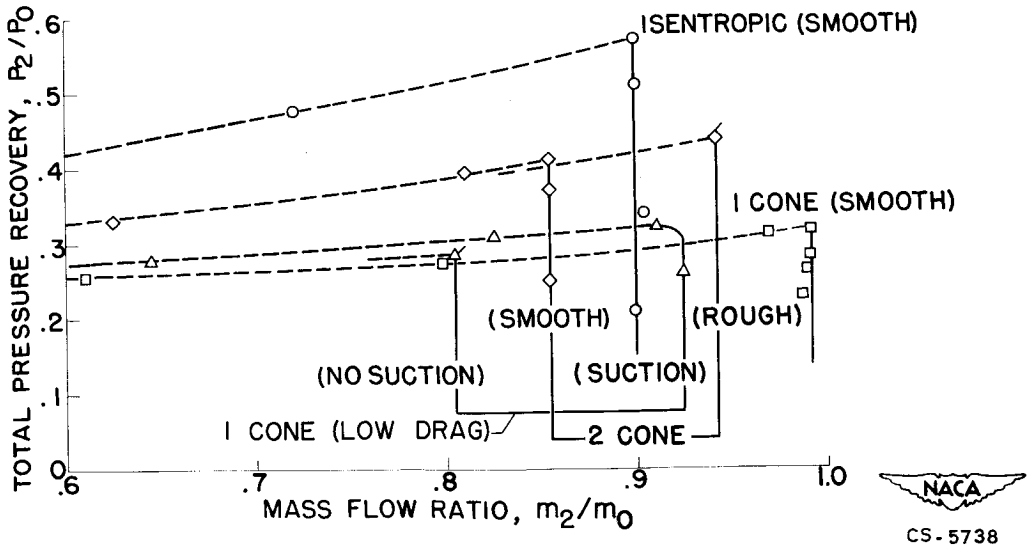

Figure 4

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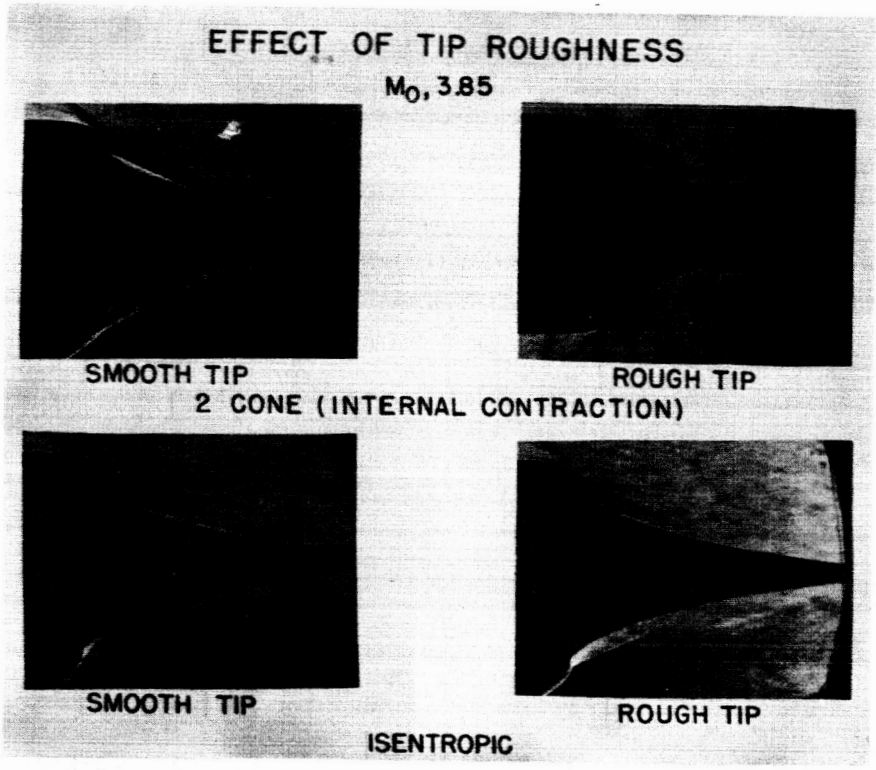

Figure 5

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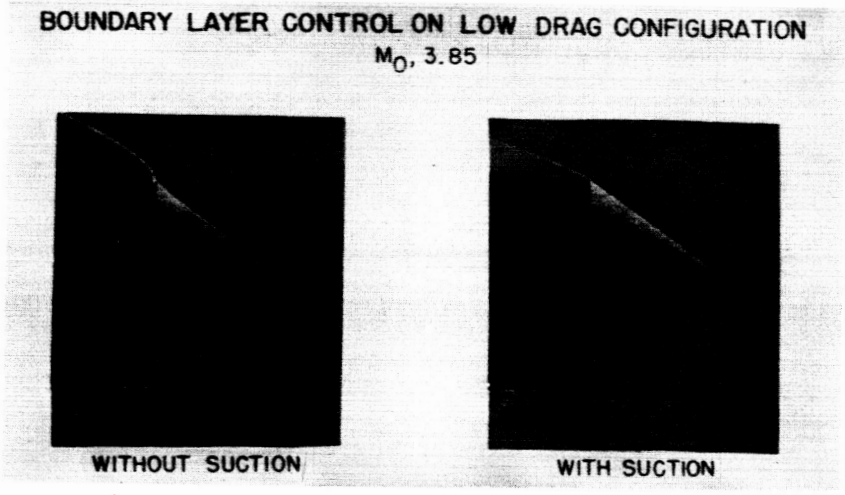

Figure 6

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TWO-DIMENSIONAL DIFFUSER CHARACTERISTICS

SPLIT-WING ISENTROPIC WEDGE

M_0, 3.85

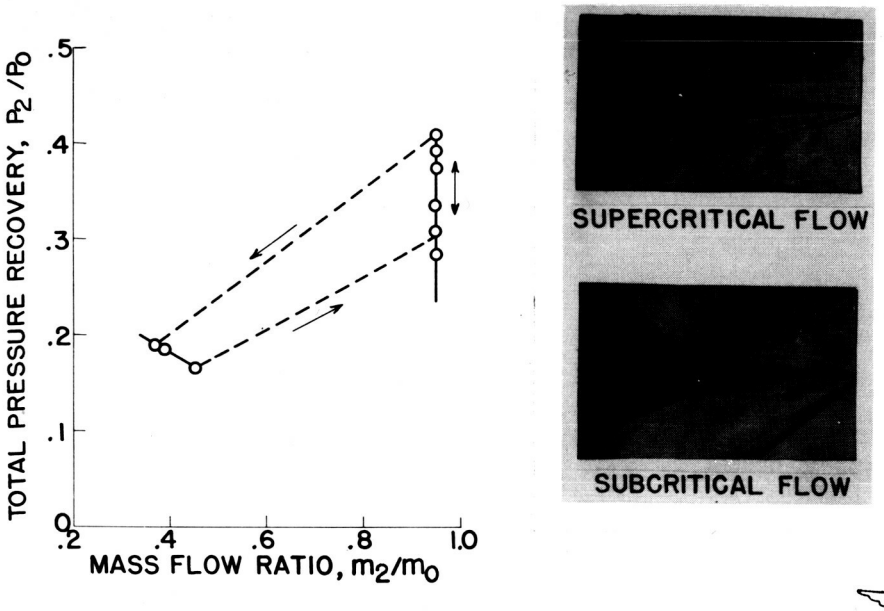

Figure 7

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EFFECT OF REYNOLDS NUMBER

M_0, 3.13

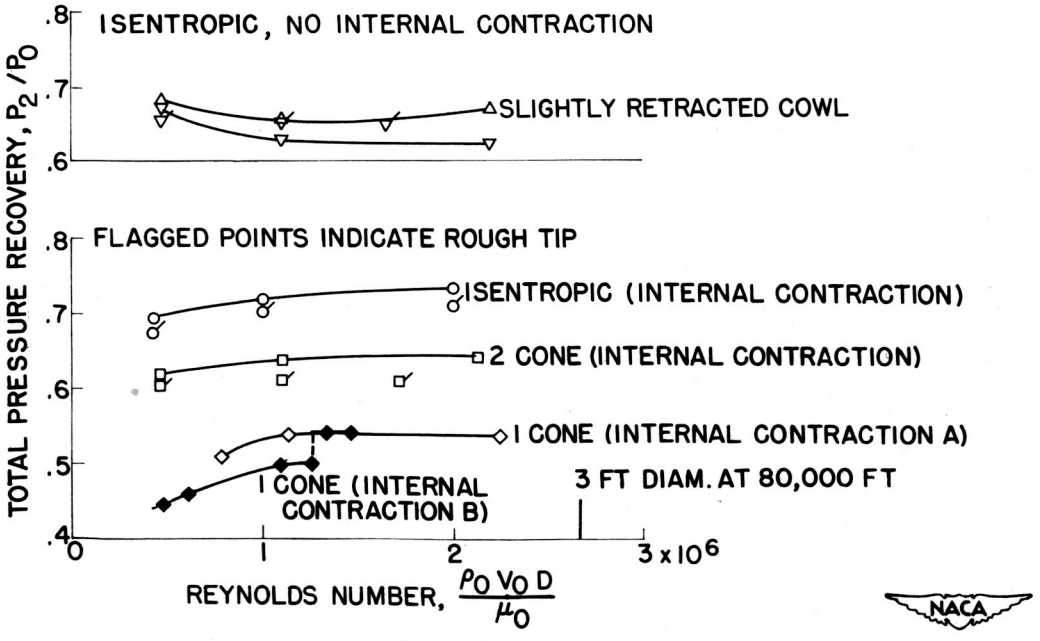

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Figure 8

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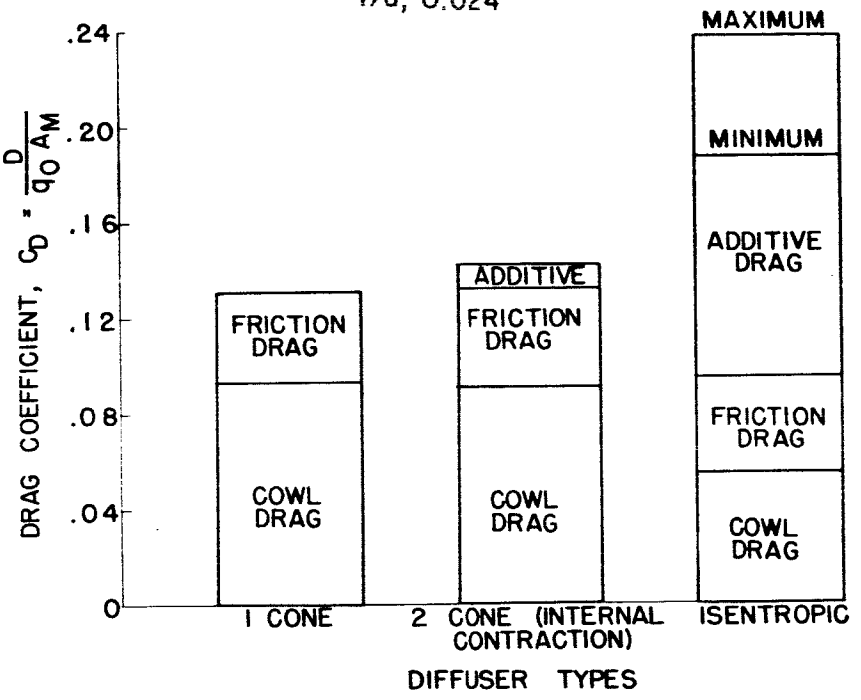
DRAGS OF RAM JETS WITH VARIOUS INLETS

M_0, 3.85
f/a, 0.024



NACA
CS-5771

Figure 9

RAM-JET ENGINES UTILIZING VARIOUS SUPERSONIC DIFFUSERS

M_0, 3.85 f/a, 0.024

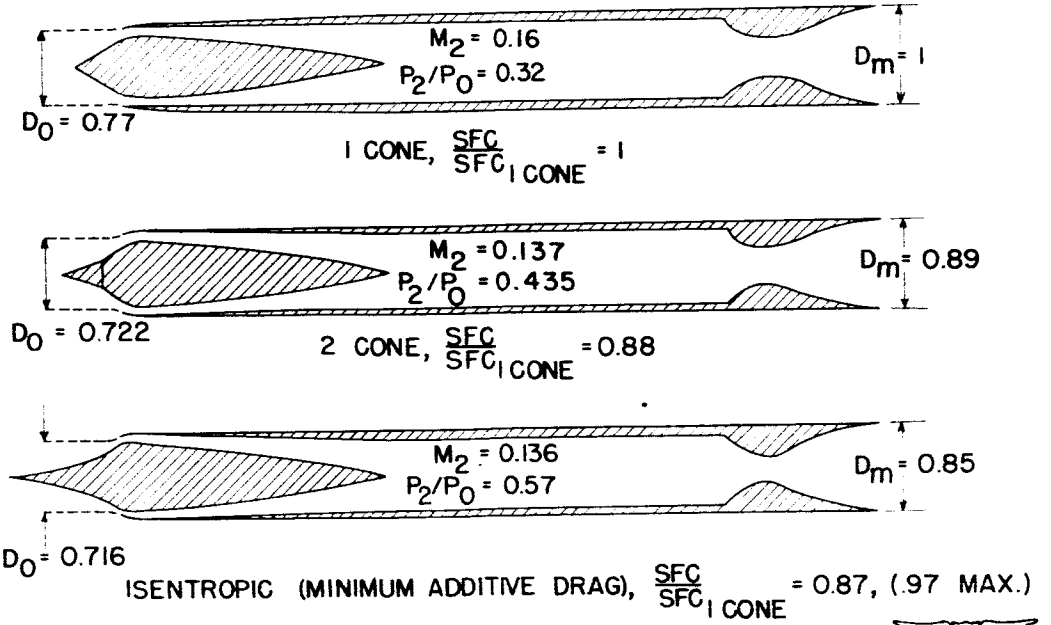


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Figure 10

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[redacted]

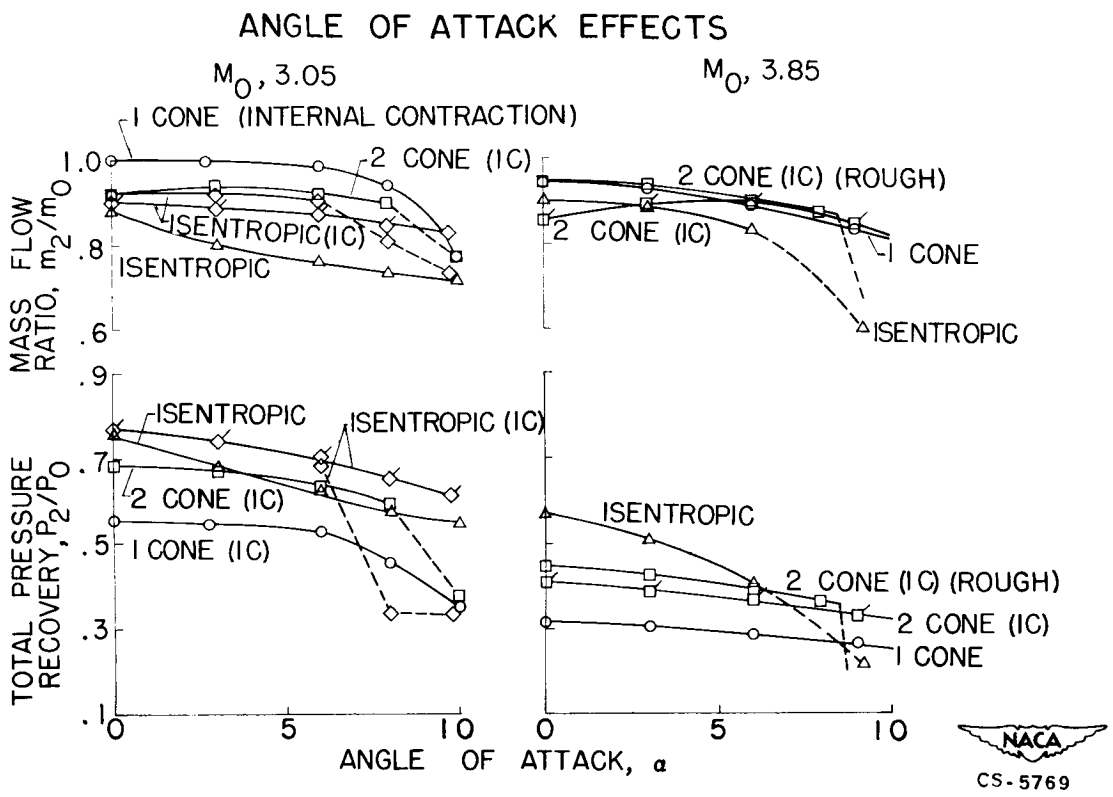

Figure 11

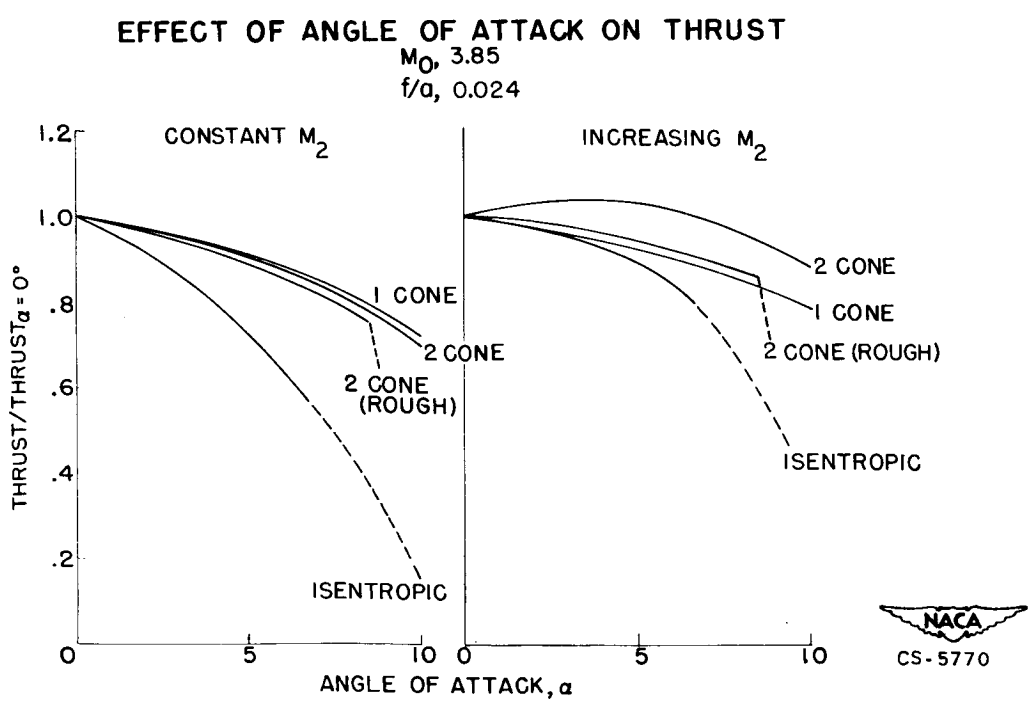

Figure 12

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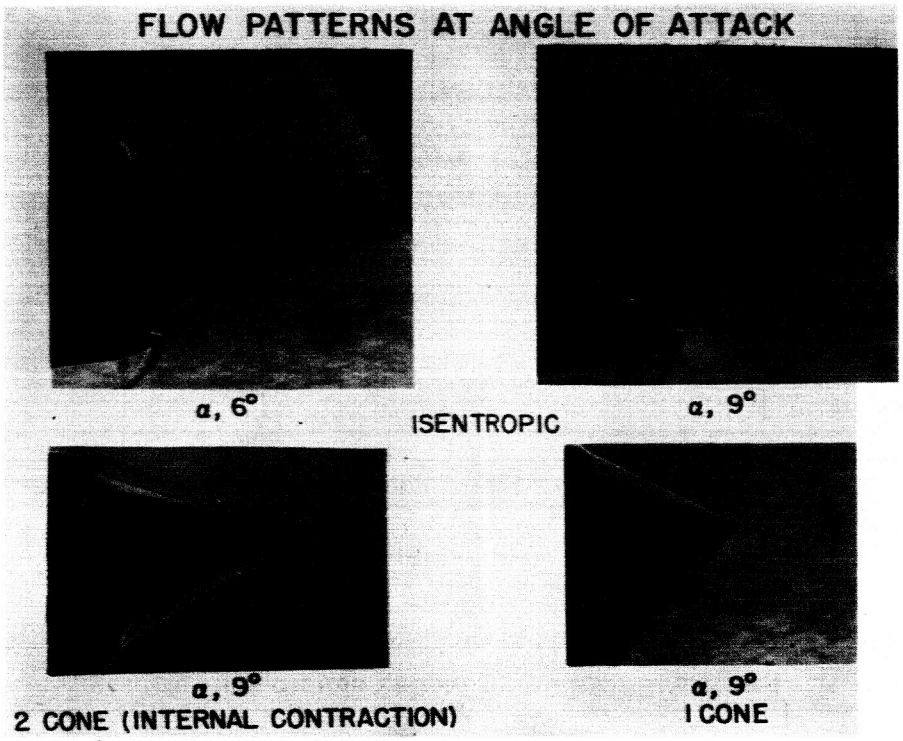

Figure 13

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SIDE INLET PRESSURE RECOVERY

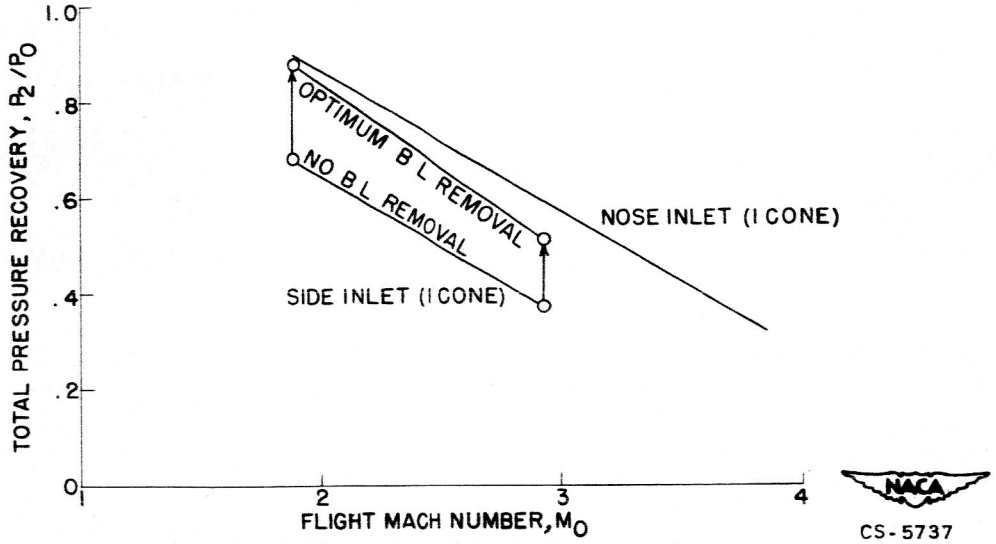

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Figure 14

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SIDE INLET THRUST

M_0, 1.88 ; δ/R, 0.15 M_0, 2.93 ; δ/R, 0.16

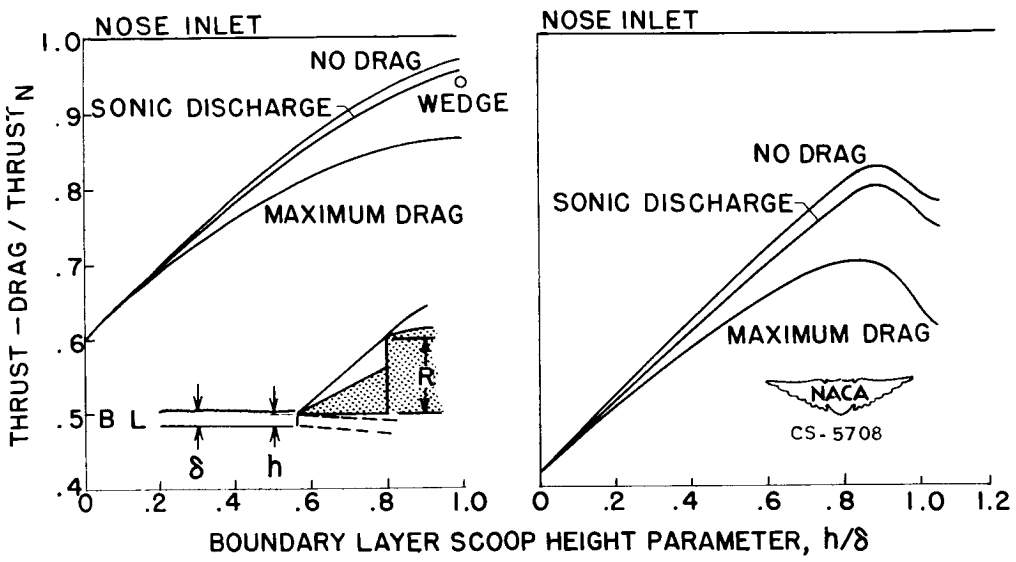

Figure 15

SIDE INLET SPECIFIC FUEL CONSUMPTION

M_0, 1.88 M_0, 2.93

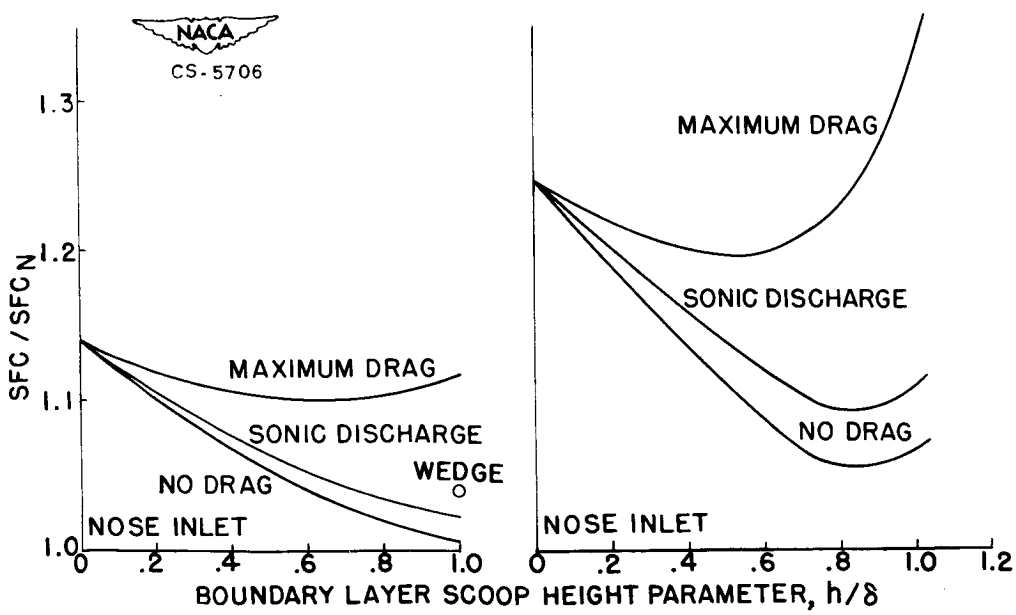

Figure 16

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11. COMMENTS IIb

[redacted]

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11. - COMMENTS IIb

By DeMarquis D. Wyatt

In the preceding paper, it has been shown that the same levels of diffusion, in terms of kinetic energy efficiency, appear to be obtainable at Mach numbers up to 4 as have been obtained at lower supersonic Mach numbers. This paper, however, discussed only the design Mach number operation of inlets. When an engine is to be used for boosting over a wide range of Mach numbers, as, for example, in the turbojet missile, the inlet performance over a range of Mach numbers becomes critical.

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The principal inlet problem over a range of Mach numbers appears to be the air flow matching problem between the inlet and the engine. Experimental data obtained at lower than design Mach numbers with a fixed-area inlet generally show the same levels of pressure recovery as those that could be achieved with on-design inlets. These pressure recoveries are, however, obtained with mass spillage ahead of the inlet with consequent drags. In addition, the inherent spillage associated with an inlet design operated at lower speeds may be greater than that required to match the engine requirements at a high pressure recovery and consequently, the engine may operate at a low pressure level with a resultant thrust reduction.

The magnitude of the matching problem is, of course, dependent on the air flow variations required by the engine. The conventional engine suffers a reduction in corrected air flow as flight speed is increased. The supersonic engine proposed by Mr. Lundin in the morning session would have a constant corrected air flow in the supersonic speed range and will create much greater inlet matching problems.

In the analytical paper opening this conference, the turbojet missiles considered were self-boosted to the cruise Mach number by conventional engines, and in the analysis it was assumed that a continuously variable inlet geometry was available such that the inlet always delivered the required engine air flow at a high pressure recovery with no drag penalty due to air flow spillage. This implies that some inlet geometry would be available in which the cowl lip area could be increased by 50 to 60 percent as flight speed is increased from sonic to the supersonic cruise speed. No mechanism by which this could be achieved is known.

In the analysis it was also assumed that the exit-nozzle throat areas and discharge areas could be independently and continuously

[redacted]

varied to give the ideal nozzle expansion ratio required at each point on the flight path. This assumption is likewise unrealistic at the present time.

The next paper will discuss the implications of the assumed inlet and outlet variations in greater detail and will show where we currently stand in relation to achieving this theoretically optimum performance.

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12. VARIABLE INLETS AND OUTLETS FOR OFF-DESIGN OPERATION

By David S. Gabriel, Roger W. Luidens

and Martin J. Saari

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12. - VARIABLE INLETS AND OUTLETS FOR OFF-DESIGN OPERATION

By David S. Gabriel, Roger W. Luidens, and
Martin J. Saari

INTRODUCTION

As pointed out in the introductory comments, the principles involved in achieving proper inlet and outlet geometry to the turbojet engine during boost are the same for any type of engine. Throughout this study, a conventionally designed turbojet engine with a take-off compressor pressure ratio of 5 is used. The missile was designed to cruise at a Mach number of 2.5 with an afterburner temperature of 2500° R.

It was necessary first to establish a missile flight plan during boost. The selection of a flight plan involves many compromises. It was found, however, that over a fairly wide range of schedules of altitude and flight Mach number, boost performance was relatively insensitive to flight plan. Accordingly, a flight plan within this insensitive range was chosen. The flight plan is shown in figure 1. The missile was boosted to a flight Mach number of 0.4 with an auxiliary rocket. Acceleration to a Mach number of 0.8 takes place at low altitude. The missile then climbs to 30,000 feet while at this Mach number and cruises for about 3000 miles. At the end of this cruise, the subsonic portion of the missile is dropped and acceleration to a Mach number of about 2.0 takes place at 30,000 feet. Climb and acceleration then continue until a cruise condition of 2.5 Mach number at 55,000 feet is reached.

In addition to the schedule of Mach numbers and altitudes, the afterburner temperature used during boost influences the performance. The afterburner temperature determines the thrust-drag margin available at each flight condition. The higher the afterburner temperature the higher the thrust margin. With high afterburner temperatures, then, the time spent in the boost is short. This time reduction, however, is achieved at the expense of high fuel consumption rates. If the afterburner temperature is reduced the time spent in boost is longer but the fuel consumption rates are correspondingly lower.

Calculations showed that if continuously variable inlets and outlets were assumed, the total fuel consumption of the missile during boost was independent of afterburner temperature over the range from 2500° to 3500° R. If the afterburner temperature was reduced much below 2500°, however, the thrust margin was eliminated and acceleration was not possible. Some afterburning was therefore required. The good boost performance over this wide range of afterburner temperatures was dependent on the assumption of continuously variable inlet and outlet

[redacted]

geometry which resulted in ample thrust-drag margins at all flight conditions. Continuously variable geometry is, of course, difficult to achieve practically. The effects on thrust of using more practical inlet and outlet arrangements are therefore examined.

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EFFECTS OF FIXED INLET AND OUTLET GEOMETRY

In figure 2, some inlet air flow characteristics are shown. The air flow taken on by the engine is plotted against the flight Mach number. Lines are shown for the air flow variation with continuously variable inlet geometry and for the air flow variation if the inlet geometry is fixed at the design point condition. The air flow with continuously variable inlet geometry reflects the engine air flow demands for the particular pressure recoveries that the inlet can supply. If the geometry is fixed at design point values, the air flow requirement of the engine and the pressure recovery that the inlet can supply interact in a complicated manner which will be explained later. The result is that at a flight Mach number of about 0.8 the air flow is reduced by about 20 percent. The pressure recoveries are reduced by about the same percentage. These large reductions in air flow and pressure recovery combine to produce large losses in thrust.

The possibility of fixing the engine outlet nozzle area at the design point condition is now considered. As shown in figure 3, additional thrust losses are to be expected. In this figure, exhaust nozzle throat area is plotted against flight Mach number. The solid line shows the throat area required to maintain a constant afterburner temperature for a continuously variable exhaust nozzle area. For any constant afterburner temperature, in this case 3500° R, the exhaust nozzle throat area must continuously decrease as flight Mach number increases in order to maintain limiting turbine outlet conditions. At the final Mach number the throat area must decrease suddenly to permit operation at the cruise afterburner temperature of 2500°. Also shown is a dashed line denoting the exhaust nozzle throat area fixed at the constant value for the cruise condition. This throat area is considerably less than the area required for an afterburner temperature of 3500°. The afterburner temperature must therefore be reduced to about 2000° R at a flight Mach number of about 0.2. This reduction in temperature results in a large thrust reduction.

The reduction in thrust associated with this temperature loss combines with the thrust reduction associated with the fixed inlet to produce the thrust variation shown in figure 4. In this figure, thrust coefficient is plotted against flight Mach number. Lines are shown for the thrust coefficients theoretically available with continuously variable inlets and outlets and with a high afterburner temperature, and for the engine equipped with fixed design point inlets and outlets.

Also shown is the missile drag coefficient. It is apparent that for continuously variable geometry a large thrust margin exists. This is the thrust margin available for acceleration. The reductions in thrust for the fixed design point geometry, however, are so great that the thrust margin becomes zero at a Mach number of about 1.8. It is evident, then, that fixed design point geometry may not be used.

These curves show that potentially a large thrust margin for acceleration exists. Some thrust losses may therefore be taken in boost without greatly affecting performance. This fact suggests the possibility of using a compromise inlet and outlet design which approaches the continuously variable case.

TWO-POSITION INLET

Next, the source of the losses with fixed geometry is investigated and some practical compromises are considered. The inlet is examined first. Figure 5 compares the flow fields for three wedge inlets at representative Mach numbers of 2.5 and 1.8. The continuously variable inlet is shown at the left. All components of this inlet are considered to be variable. The wedge angle for best pressure recovery is selected at each flight condition and the inlet cowl intersects the oblique shock. The cowl diameter is varied to match exactly the engine air flow requirement. The normal shock is always at the lip and the inlet operates at all flight conditions without additive drag. The inlet fixed at the design point, shown here in the second column, is exactly the same as the continuously variable inlet at the final Mach number of 2.5. As the Mach number decreases, however, the oblique shock angle increases and considerable spillage occurs. At a Mach number of 1.8 the oblique shock has detached from the wedge and forms a bow wave. Under these conditions the air flow that can enter the inlet is limited by choking the throat. Expansion to supersonic velocity then occurs in the subsonic diffuser. A shock system forms which reduces the pressure at the compressor inlet to match the reduced air flow. It is this reduction in air flow and pressure recovery which is responsible for a large part of the engine performance reductions previously shown. A compromise inlet, which reduces the air flow and pressure recovery losses associated with the fixed design point inlet, is shown in the final sketches. This inlet has a reduced wedge angle of 14° but the same cowl diameter as the design point inlet. With these conditions the air flow and pressure recovery losses experienced by the design point inlet are largely eliminated. The necessity for changing the wedge angle at a flight Mach number of 2.1 is indicated by the top figure. At Mach numbers above 2.1 the oblique shock and the bow shock for the wedge angle of 13° intersect at a position within the cowl diameter. Several investigations have shown that this condition is

conducive to diffuser instability and to low pressure recoveries. These undesirable conditions may be avoided by increasing the wedge angle to 20° at a flight Mach number of 2.1.

A two-position inlet then results. The compromise inlet with a 14° wedge is used throughout the Mach number range up to 2.1. At 2.1 the wedge angle is increased to 20°. The cowl diameter is constant throughout the flight path. The inlet is then used in this fixed condition for flight Mach numbers from 2.1 to 2.5. Similar two-position inlet arrangements can be derived for spike-type inlets by using a simple translation of a spike.

The performance of the compromise two-position inlet is compared with the performance of the continuously variable inlet in figure 6. Plotted here are the pressure recovery ratios and induced drag as a function of flight Mach number. The solid lines show the performance calculated for the continuously variable inlet and the dashed lines, the performance of the compromise two-position inlet. Some losses in pressure recovery are taken at low flight speeds. At the flight Mach number of 2.1 the wedge angle is increased to 20° causing this sudden drop in pressure recovery. As the flight Mach number is increased above 2.1 the pressure recovery approaches the continuously variable case, until at the final flight Mach number of 2.5 design point performance is restored.

The air spillage for the compromise design results in the additive drag coefficients as shown by the lower dashed curve. This additive drag, of course, reduces the available thrust margin but to a much smaller degree than the use of the fixed design point inlet. These additive drags could be largely avoided by the use of a cowl extension for the lower Mach number flight. This extension would consist of simply a segment of inlet cowl extended along the inlet streamlines at low flight speeds. The presence of this cowl extension would be undesirable at high flight speeds. The cowl extension then could be simply blown off at a Mach number around 2.0. Some wind tunnel tests of a blow-off cowl have been made at this laboratory with a view to solving some of the inlet parting problems. These preliminary tests have shown that many difficult problems associated with separation must be solved before such blow-off cowls may be practically used.

VARIABLE AREA EXHAUST NOZZLE

It has been shown that acceleration is not possible with fixed design point geometry. However, good inlet performance, approaching that theoretically available with continuously variable geometry, may be achieved by using a simple two-position inlet arrangement. Consideration is now given to the exhaust nozzle.

At high flight Mach numbers the pressure ratio across the exhaust nozzle is very high. For example, at a flight Mach number of 2.5 the nozzle pressure ratio is over 16. For maximum nozzle efficiency under these conditions a convergent-divergent nozzle is required. At lower flight speeds the nozzle pressure ratio is considerably reduced. The exhaust nozzle then must operate over a wide range of pressure ratios. The effect of pressure ratio on the performance of several fixed convergent-divergent nozzles is demonstrated in figure 7. The design of the nozzles is indicated by the sketch at the top of the figure. The nozzles were constructed of simple conical elements. Experimental data are plotted as the ratio of measured jet thrust to theoretical jet thrust for complete isentropic expansion against the nozzle pressure ratio. Lines are shown for four different nozzles having expansion ratios (ratio of outlet area to throat area) varying from 1.13 to 2.65. A line is also shown for a convergent nozzle. For any given convergent-divergent nozzle maximum nozzle efficiency is obtained at only one pressure ratio. The pressure ratio for maximum efficiency is very close to the theoretically correct values. If the nozzle is operated at pressure ratios much less than the design point values, considerable overexpansion takes place. The shock losses associated with this overexpansion result in large efficiency losses as indicated by the curves.

Because of facility limitations, data on the low expansion ratio nozzles were not obtained at very high pressure ratios. The curves would be expected to approximately parallel the convergent nozzle curves. Comparison of the peak thrust ratios with the convergent nozzle line shows that for pressure ratios above 4 an appreciable improvement in nozzle efficiency may be realized by using a convergent-divergent nozzle at its design point.

The large losses in thrust at off-design points, however, indicate that for the turbojet engine exhaust nozzle, which must operate over a wide range of pressure ratios, a nozzle with varying expansion ratio is desirable. Accordingly, a variable area nozzle was designed and tested recently at this laboratory. The nozzle performance is described in figure 8.

A drawing of the nozzle is shown at the top of the figure. This nozzle was also constructed of simple conical elements. The throat area was varied by translating the external shell as indicated by the dashed lines. It is characteristic of any plug-type nozzle with constant inlet and outlet diameters that for any given throat area the nozzle can have only one expansion ratio. In other words, if the throat area is chosen, no independent setting of expansion is possible. The performance of the nozzle is shown in a plot of actual to theoretical jet thrust ratio against nozzle pressure ratio. Lines are shown for each of three fixed positions of the nozzle, corresponding to expansion ratios of 1.5 to 2.5. At each fixed position, the nozzle exhibits characteristics typical

of fixed convergent-divergent nozzles, as shown in the previous figure. However, the maximum efficiency of the nozzle is the upper envelope of the various curves. The dashed line is the locus of maximum efficiency points from the fixed nozzle data of figure 7. By comparing the dashed line with the upper limits of the data curves it can be seen that the variable area exhaust nozzle has efficiencies very near the best that were obtained with fixed nozzles over a range of pressure ratios from about 7 to 20. The variable area nozzle then appears to be admirably suited to the turbojet engine application.

The matching characteristics of this variable area nozzle to a turbojet engine are illustrated in figure 9, where exhaust nozzle throat area and expansion ratio are plotted against flight Mach number. The exhaust nozzle throat areas shown are the areas required to maintain limiting turbine outlet conditions with a constant afterburner temperature of 3500°. As previously stated, the throat area must continuously decrease as flight Mach number increases. It is possible with the experimental variable area nozzle to duplicate exactly this throat area requirement. If the exhaust nozzle were to be completely expanded, the expansion ratio variation indicated by the dashed line below would be required. As previously indicated, independent control of throat area and expansion ratio for the given variable area nozzle is not possible. The variation in expansion ratio which should result from this schedule of throat areas is shown by the solid line. As indicated by the curves, the nozzle would be considerably overexpanded at low flight Mach numbers and somewhat underexpanded at high flight Mach numbers.

The resulting variations in nozzle efficiency and thrust ratio are shown in figure 10. In this figure, actual to theoretical thrust ratio is plotted against flight Mach number for the particular schedule of throat areas and expansion ratios shown in figure 9. It is apparent that although the losses at high flight Mach numbers are not large, the losses at low flight Mach numbers are appreciable. These losses would not be serious during boost where large thrust margins are available; however, if the missile were designed to cruise appreciable distances at subsonic speeds, losses of this magnitude would become intolerable. The exhaust nozzle during subsonic cruise must be converted to a convergent nozzle. This conversion may be accomplished by adding a convergent fairing to the outlet of the exhaust nozzle. This convergent fairing then would convert the exhaust nozzle to a convergent nozzle for the portion of the flight up to a Mach number of 0.8 and including the subsonic cruise. When acceleration was started, the convergent fairing could be ejected and the variable area nozzle would be used during acceleration as previously explained.

TWO-STAGE BURNER

Although the variable area exhaust nozzle is an adequate device for this application, the presence of the plug in the hot gas stream creates

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some difficult problems. The plug must be either cooled or fabricated of heat-resistant materials. An ingenious device for avoiding these problems is a two-stage burner shown in figure 11. With this device it is possible to effectively vary the exhaust nozzle area and thereby to produce high afterburner temperatures without the complication of moving parts. At the top of figure 11 is shown a sketch of the conventional afterburner with convergent-divergent exhaust nozzles. This burner and nozzle are designed for the cruise condition. If the missile is in flight at a sufficiently high speed, the pressure ratio across the exhaust nozzle is large enough to produce supersonic velocities at the nozzle outlet. Suppose a second-stage burner and exhaust nozzle are added as shown in the second sketch. The second-stage exhaust nozzle throat is larger than the first stage. If no burning is done in the second stage, supersonic flow will be established through the second-stage burner and nozzle in the same manner as for a supersonic wind tunnel.

If fuel is then injected and burned in the supersonic stream within the second-stage burner, the Mach number in the second throat will decrease. The first throat will remain choked and the first-stage burner will be unaffected. Additional fuel could then be burned in the second stage until the second throat chokes. At this condition, both throats are choked. A further increase in fuel flow to the second stage will result in establishment of a shock system in the outlet of the first-stage nozzle as shown by the third sketch. Additional fuel may then be burned in the second stage until the shock so established has moved back to the throat of the first nozzle, as shown in the fourth sketch. All this may be done without influencing the first-stage burner. By using the second-stage burner, then, the final afterburner temperature can be considerably increased while limiting turbine outlet conditions are maintained.

The two-stage burner could be used during boost to achieve higher afterburner temperatures. At the end of boost, the second stage would be dropped and the afterburner designed for the cruise condition would remain on the engine.

A two-stage burner has recently been operated here experimentally. This burner was designed for a ram-jet engine but the results serve equally as well to illustrate its operation on a turbojet afterburner. The performance of the experimental burner is shown in figure 12. Thrust coefficient is plotted against over-all temperature ratio. Points are shown for operation with burning in the first stage only and with burning in both stages. With first-stage burning only, thrust was increased up to the value shown by the last circled point by increasing temperature ratio across the first stage. At this point rated burner inlet conditions, corresponding to rated turbine outlet conditions in the turbojet case, were established. The fuel-air ratio in the

first-stage burner was maintained constant from that point. Additional thrust was then obtained by burning in the second stage. It is of interest to note that these data are an example of successful burning in a supersonic stream. The path traversed by the square points corresponds to the process of burning in the supersonic stream, establishment of the shock between the throats, and at the final point location of the shock at the first throat. In the complete range of operation from the last circle to the last square an increase in thrust of almost 2 to 1 was accomplished without influencing the first-stage burner inlet conditions.

It is apparent from these preliminary tests that the two-stage burner does have some interesting possibilities for application to the turbojet engine during boost. Consideration is now given to matching this device to the engine during boost. Figure 13 shows the second throat area plotted against flight Mach number during boost. The dashed line is the variation in throat area ideally required to maintain a constant afterburner temperature of 3500° R. To avoid the complication of moving parts a fixed exhaust nozzle should be used at the outlet of the second-stage burner. A nozzle throat area corresponding to the solid line may be used. The throat area is less than the area required for 3500° at low flight speeds and greater at high flight speeds. The maximum afterburner temperature that can be obtained with the two-stage burner then varies from about 3000° to 3800°. Some thrust reduction must therefore be taken at low flight speeds. At the final Mach number the second-stage burner and nozzle are dropped and the cruise condition is obtained.

It is evident that the throat area of the second-stage nozzle is fixed by the afterburner temperature requirements. A nozzle expansion ratio may still be chosen, however. The selection of expansion ratio is dependent on nozzle efficiency, as will be shown in figure 14.

Some of the data for fixed area convergent-divergent nozzles previously shown in figure 7 are replotted here. The ratio of actual to theoretical thrust is plotted against flight Mach number. Lines are shown for three expansion ratios. The second-stage nozzle will not be used during cruise. An expansion ratio which results in the best compromise nozzle efficiency during boost is therefore selected. As shown in figure 13, the afterburner temperature, and therefore the thrust, is low at low flight speeds and high at high flight speeds. The nozzle expansion ratio then should be chosen to supply best performance at low speeds. As indicated by the curves an expansion ratio of about 1.30 yields the desired characteristics of high efficiency at low speeds with moderate losses at high speeds.

PERFORMANCE WITH TWO-POSITION INLET AND VARIABLE AREA EXHAUST NOZZLE

Up to this point it has been shown that the inlet and outlet geometry requirements of the engine during boost may be closely approached by the use of two-position inlets and a variable area exhaust nozzle or a two-stage afterburner. Some losses were unavoidable with these devices. The improvement of the thrust margin over the fixed geometry case is now considered. Figure 15 shows the thrust coefficient plotted against flight Mach number. The missile drag coefficient is shown as well as the thrust with fixed design point geometry. Plotted at the top is the thrust curve obtained with two-position inlet geometry and the variable area exhaust nozzle.

It is evident that a large increase in thrust has been realized. An appreciable thrust margin now exists at all flight conditions. Although an ample thrust margin is available, fuel consumption rates have been increased during boost. The question then remains as to what degree the increased fuel consumptions incurred in using the more practical geometry arrangements influence the missile performance. This question is answered for the two-stage turbojet missile in figure 16. Missile gross weight is plotted against range. A line is shown for the missile weight using the continuously variable inlet geometry as discussed in "Analysis of Missile Propulsion Systems" and for the missile weight if two-position geometry is used. A slight increase in missile weight of about 2 or 4 percent results from using the simpler geometry.

SUMMARY

In summary, it has been shown that reasonably good boost performance, approaching that theoretically available, can be obtained by the use of simple inlet and outlet arrangements if careful consideration is given to the selection and matching of these components. Expectation is far from realization, however, and before this good performance may be actually achieved much knowledge must be gained and many practical difficulties must be overcome.

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BOOST FLIGHT PATH FOR TURBOJET MISSILE

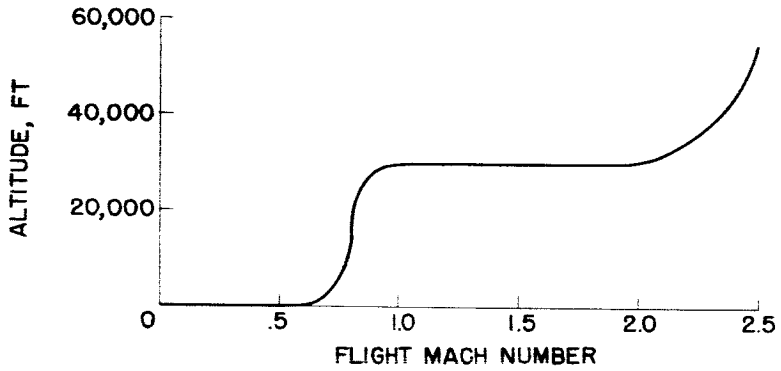

Figure 1

AIR FLOW CHARACTERISTICS OF TURBOJET INLETS

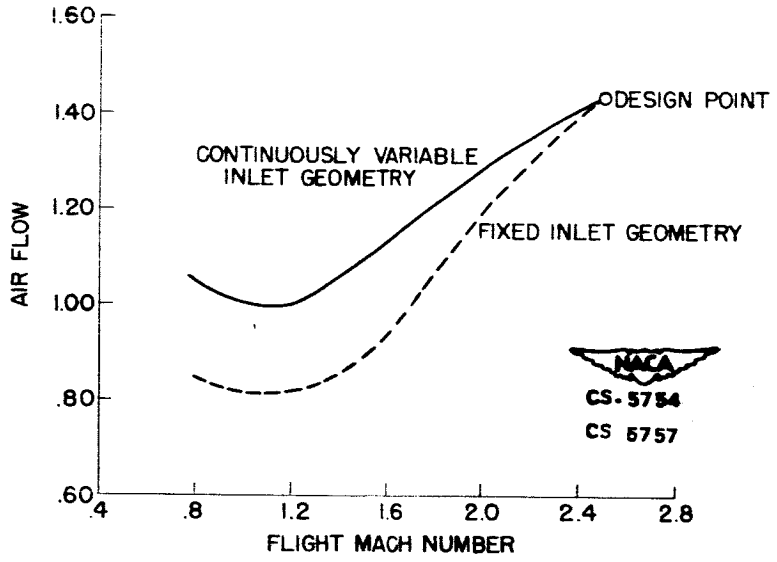

Figure 2

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AFTERBURNER OUTLET NOZZLE THROAT AREAS

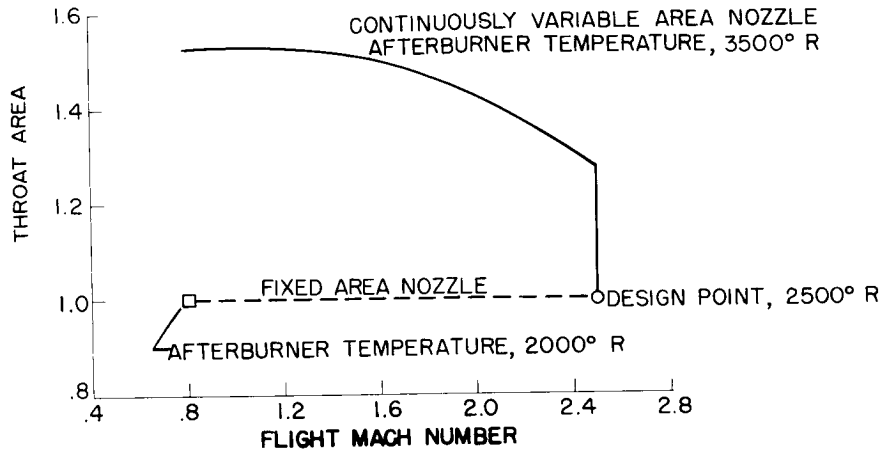

Figure 3

EFFECT OF INLET AND OUTLET GEOMETRY ON TURBOJET THRUST

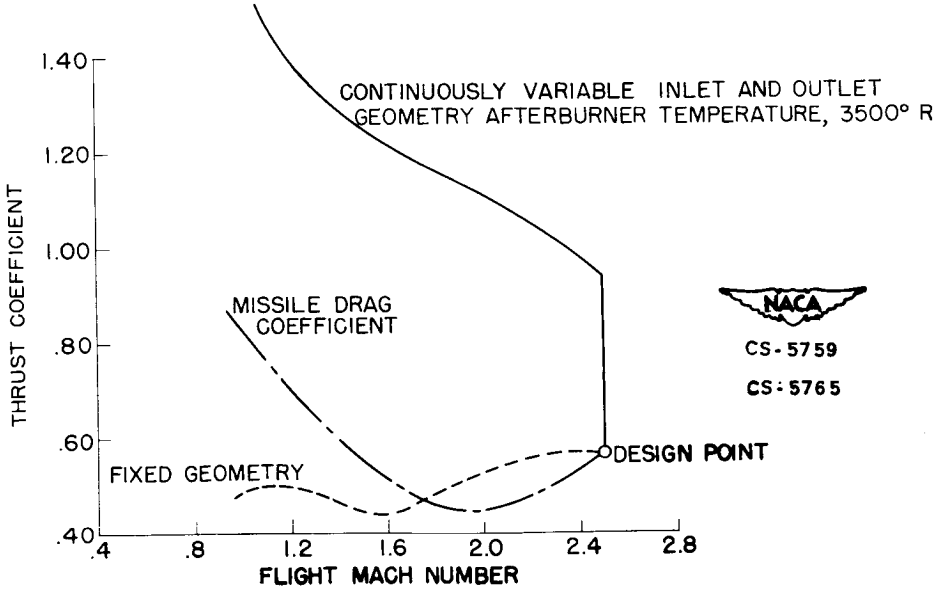

Figure 4

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TURBOJET WEDGE INLETS

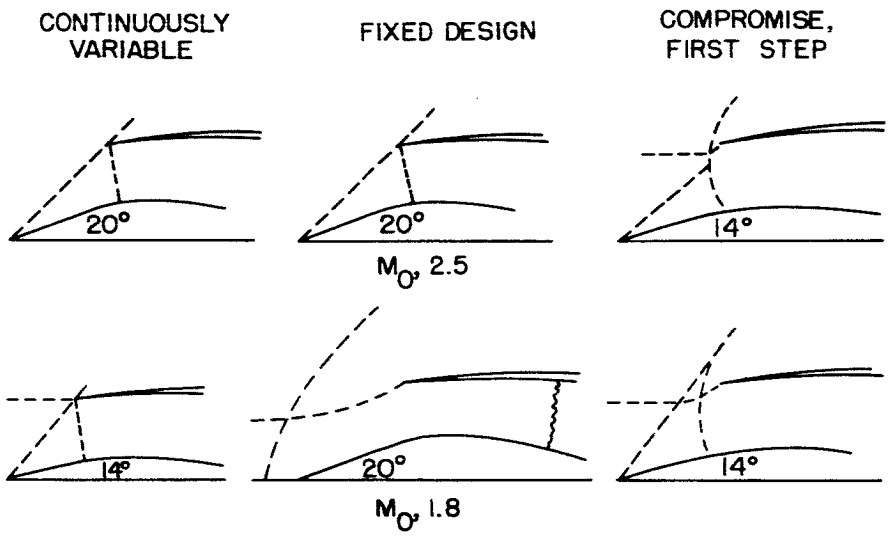

Figure 5

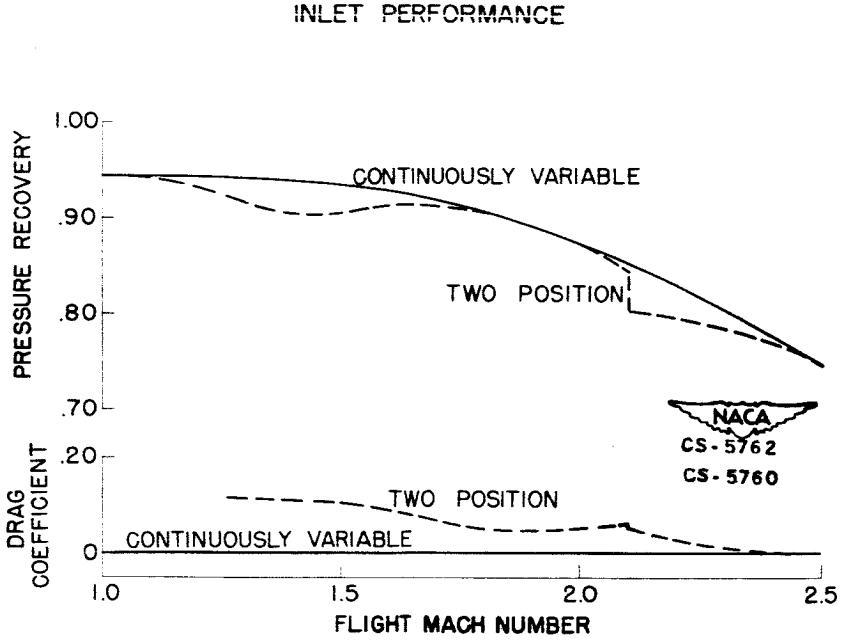

Figure 6

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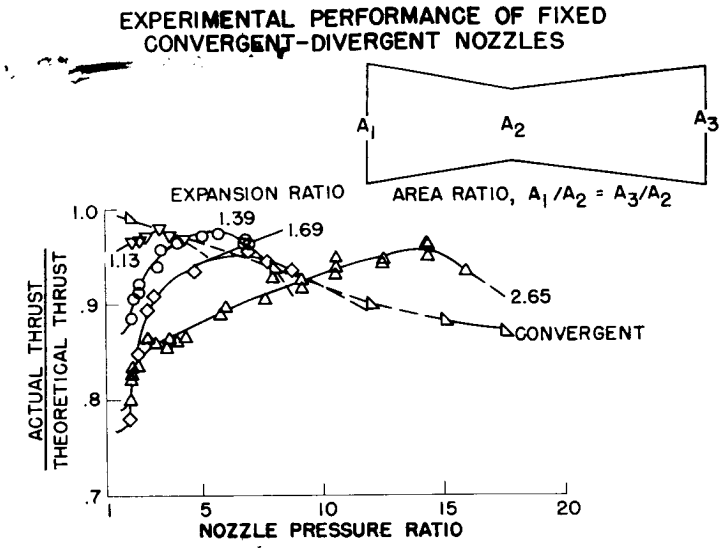

Figure 7

EXPERIMENTAL PERFORMANCE OF VARIABLE AREA NOZZLE

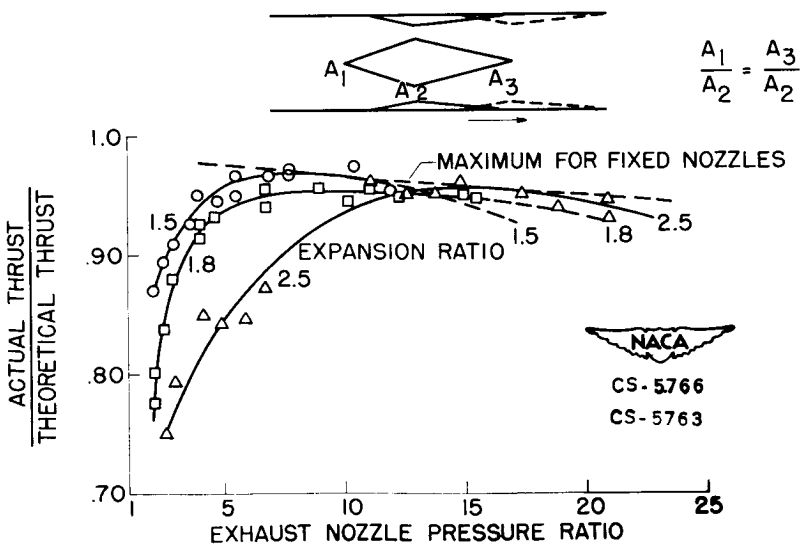

Figure 8

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MATCHING OF VARIABLE AREA NOZZLE TO TURBOJET ENGINE

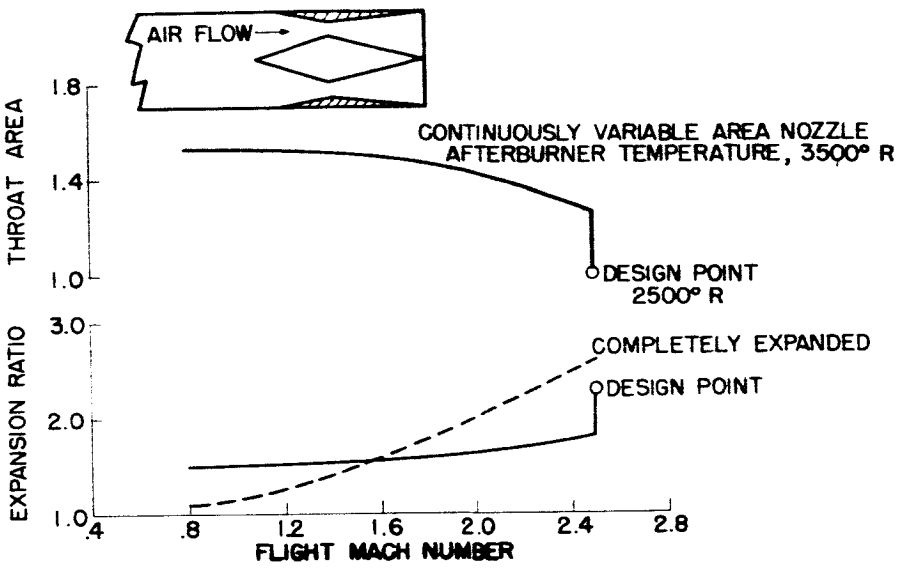

Figure 9

VARIABLE AREA NOZZLE EFFICIENCY

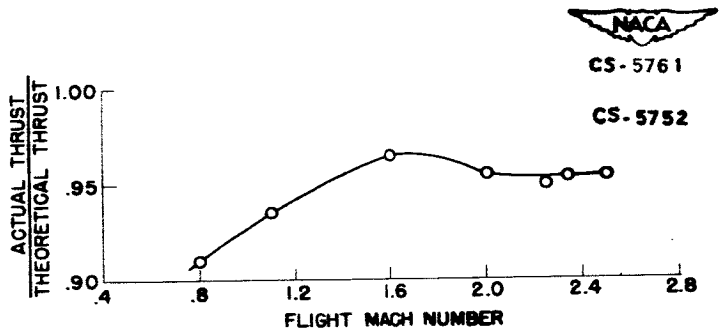

Figure 10

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2-STAGE AFTERBURNER

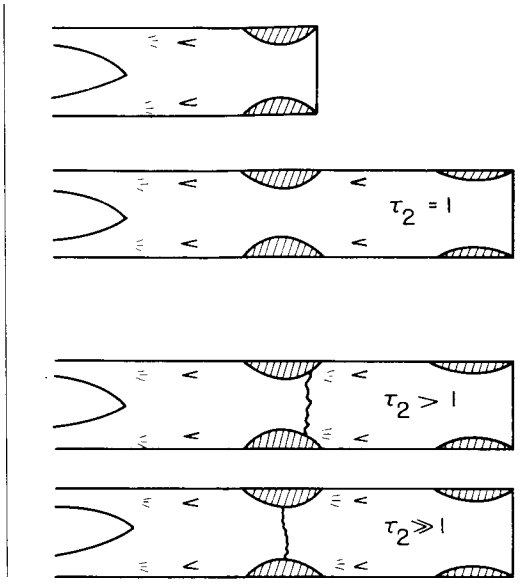

Figure 11

EXPERIMENTAL PERFORMANCE OF 2-STAGE AFTERBURNER
RATIO OF THROAT AREAS, 1.7

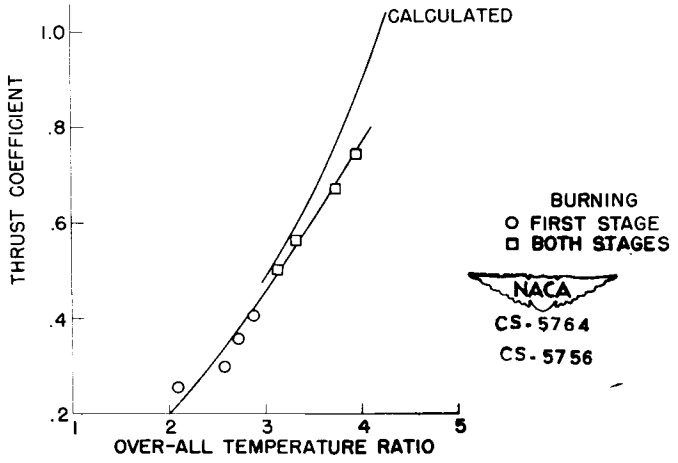

Figure 12

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MATCHING 2-STAGE AFTERBURNER TO ENGINE

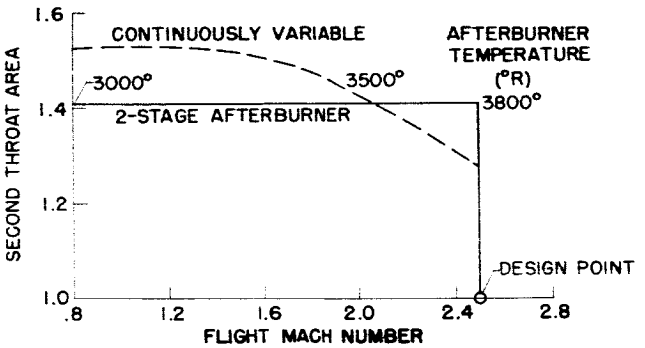

Figure 13

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EFFECT OF EXPANSION RATIO ON SECOND-STAGE NOZZLE PERFORMANCE FOR 2-STAGE AFTERBURNER

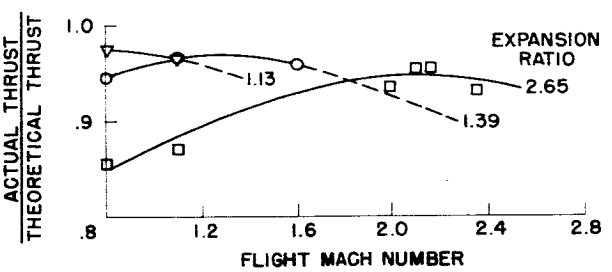

Figure 14

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EFFECT OF INLET AND OUTLET GEOMETRY ON TURBOJET THRUST

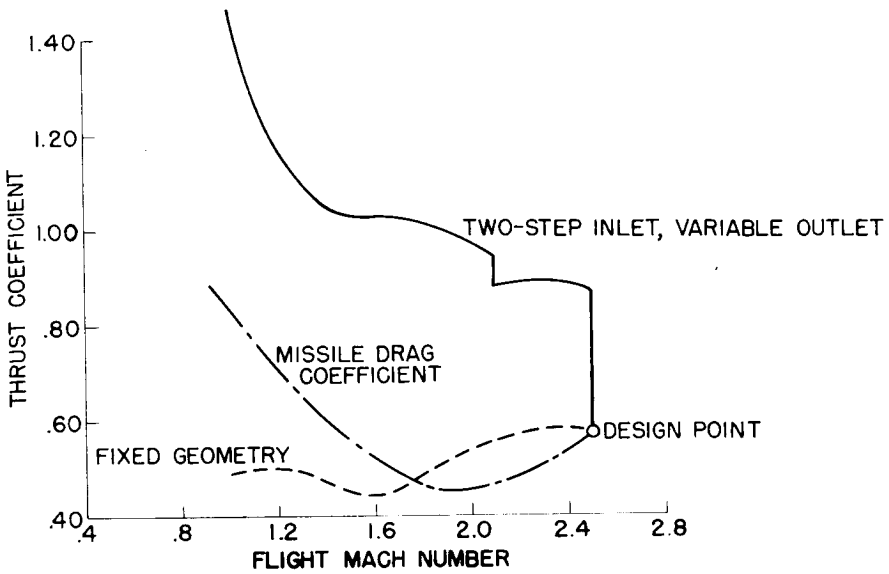

Figure 15

EFFECT OF INLET AND OUTLET GEOMETRY ON GROSS WEIGHT OF 2-STAGE TURBOJET MISSILE

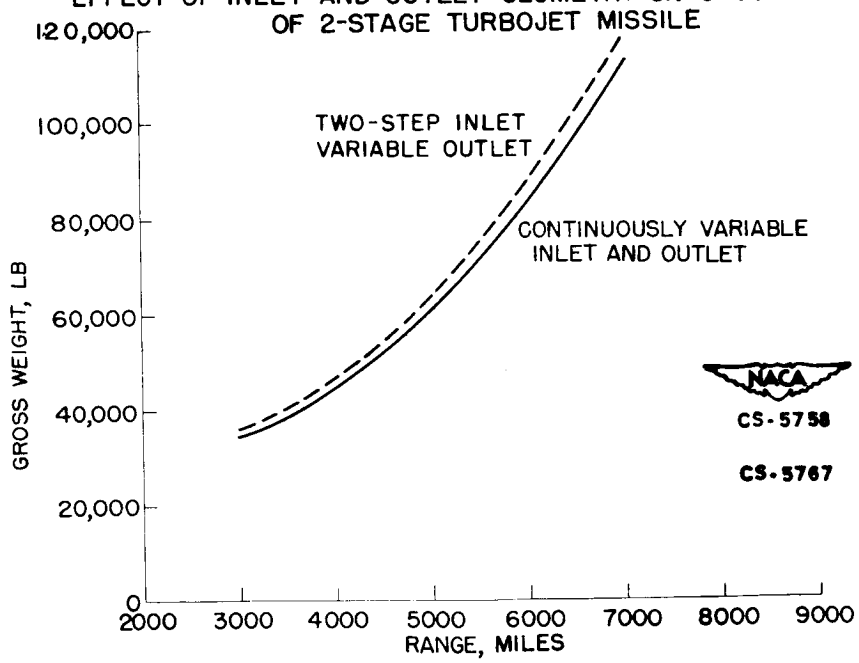

Figure 16

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13. COMMENTS IIc

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13. COMMENTS Etc.

By DeMarquis D. Wyatt

In the preceding papers the effect of various engine design parameters on the initial weight requirements of long-range supersonic missiles has been studied. From these analyses, it was found that the initial choice of some of these variables is relatively unimportant, whereas certain others critically affect the missile. In almost every case, however, once a design parameter is removed from a variable category to a fixed quantity in the engine, the predicated performance of the resultant missile design is extremely dependent on obtaining the assumed component performance.

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If the assumptions regarding the potential component performance levels are valid, then the success or failure of the power-plant phase of missile flight largely becomes the responsibility of the engine control system as measured by its capacity to maintain individual components at their assumed operating conditions in the face of transients resulting from external disturbances.

Depending on the flight path and the powerplant, the control system requirements may be relatively simple or extremely complex. For example, a ram-jet missile rocket boosted to the cruise Mach number will require only that the engine control maintain a constant engine operating condition through fuel-flow control. On the other hand, the use of a turbojet engine of the type described by Mr. Lundin to boost as well as cruise the missile will require the control system to follow some schedule during boost in which a number of engine parameters must be simultaneously varied.

Control system requirements are discussed in greater detail in the next paper and the nature of the systems required for different engine types is shown.

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14. CONTROLS FOR MISSILE PROPULSION SYSTEMS

By Seymour C. Himmel

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14. - CONTROLS FOR MISSILE PROPULSION SYSTEMS

By S. C. Himmel

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The control of propulsion systems can be considered to comprise two major elements: (1) attainment of desired engine performance throughout a flight plan, and (2) minimization of departures from desired performance during transients resulting from external disturbances acting on the engine aircraft system. This latter element, which exists for all aircraft, assumes greater importance at supersonic flight speeds because engine performance is greatly affected by flight velocity in this region.

This paper will discuss some of the engine control problems associated with the boost and cruise phases of missiles propelled by air-consuming power plants. Methods for obtaining required engine performance and for stabilizing missile flight velocity are presented for these systems; the principles of operation of these controls, rather than the detailed design considerations, are discussed.

The control problems associated with the ram-jet engine are considered first. The basic problems of engine controls for a ram jet operating at supersonic flight speeds are primarily the result of the characteristics of the inlet diffuser. These characteristics are such that departure from the design point results in either reduced thrust and efficiency or unstable engine operation. The control problem for the ram jet resolves itself, therefore, into the determination of the method of manipulation of the engine input variables to attain satisfactory diffuser and, hence, engine performance.

In order to achieve desired range with minimum missile weight with a ram-jet propulsion system, it is necessary to operate the engine at its point of minimum specific fuel consumption. For a ram-jet engine this occurs at the point of critical diffuser flow. This condition establishes the engine control requirement, which is to keep the engine at its critical operating point.

Although the fixed-geometry ram jet is essentially a fixed-flight-speed device, consideration must be given to engine operation at speeds other than the design speed because the missile may introduce disturbances in engine operation by deviating from the required flight conditions, in particular, changes in flight velocity that greatly affect inlet performance.

The fixed-geometry ram jet has but one input variable which can be manipulated to obtain the desired engine operation; this variable is engine fuel flow. The effective fuel flow required for critical engine

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operation is shown in figure 1 as a function of flight Mach number for a fixed-geometry ram jet. The curve shown is for an engine designed to cruise at a flight Mach number of 3.0. The fuel flow required for critical engine operation is seen to increase exponentially with flight speed and varies three-fold over the range of Mach numbers shown. In order to maintain critical operation of the engine it is, of course, possible to schedule the fuel flow with Mach number, but this requires accurate knowledge of burner performance, especially with regard to combustion efficiency, and such a control scheme can offer no assurance of proper engine operation.

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It is advantageous to utilize some engine variable which is truly indicative of the engine operating point to control the fuel flow to the engine. Two engine parameters which satisfy this requirement and are not functions of combustion efficiency are the diffuser-exit Mach number and the ratio of diffuser-exit total pressure to ambient static pressure. The variation of these parameters with flight Mach number required for critical engine operation is shown in figure 2. The diffuser-exit Mach number is a gradually decreasing function of flight Mach number, varying from 0.21 to 0.186 at Mach numbers of 2.6 and 3.2, respectively. The diffuser pressure ratio for critical operation has a rising characteristic varying from 12.6 to 24.2 at Mach numbers of 2.6 and 3.2, respectively. These curves represent, in fact, possible control relations for critical ram-jet operation which can be maintained by variations of engine fuel flow. That is, the desired value of either of these two parameters can be used as scheduled functions of the flight Mach number to be attained by manipulation of the fuel flow in a closed-loop fuel control. The application of each of these variables in such a control system is now considered.

In figure 3 is shown a block diagram of a possible control system utilizing the control relation between pressure ratio of the diffuser and flight Mach number. In this system the flight Mach number is measured by means of a Machmeter or similar instrument and this signal is fed into a scheduler. This scheduler has built into it a computing device (such as cams or specially wound potentiometers) such that its output signal bears the functional relation to flight Mach number indicated in figure 2. If operation only at a fixed Mach number is desired, a fixed reference signal can be used. In either event, the output of the scheduler is the required value of diffuser pressure ratio P_2/p_0. The value of P_2/p_0 existing in the diffuser exit is measured and this signal is fed back to be compared with the required value of P_2/p_0 in an error detector. The output of this detector is the difference between the required values and the existing value of P_2/p_0. If a difference exists between these values, this signal is fed into a fuel control which acts to increase or decrease engine fuel flow

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as required, thus bringing the error to zero and attaining the desired operating point. This system is advantageous because it automatically corrects for the variation in ambient pressure with altitude, and it is not influenced by variations in combustion efficiency.

This system has, however, one important disadvantage: such a control may easily be "fooled" into placing the engine in the subcritical regime with possibly attendant diffuser pulsing and thrust loss. How this can occur can be deduced from figure 4 in which is shown a plot of P_2/p_0 against effective corrected fuel flow ηW_f at a constant flight Mach number. The curve has been subdivided into supercritical and subcritical operating regions. In the supercritical engine operation regime, it is seen that an increase in fuel flow results in an increase of P_2/p_0; therefore, if the control shown in figure 3 senses a P_2/p_0 less than the set value, it will act to increase the fuel flow to raise P_2/p_0, thus bringing the engine towards the critical point. At the critical point, there is a discontinuity in the pressure recovery curve; the nature of this discontinuity is a function of the diffuser characteristics. In general, there are two possible behaviors of P_2/p_0 in the subcritical region: P_2/p_0 may have a constant value, or P_2/p_0 may decrease with increasing fuel flow with possible attendant pulsing. In either event it is possible for the control to increase the fuel flow continuously in an effort to raise the pressure ratio, and engine failure eventually occurs. In the case of the flat diffuser characteristic, this could be caused by an error in the control schedule, for example, a calibration error such that a value of P_2/p_0 higher than can be attained at a given Mach number is set. For a diffuser with a decreasing pressure ratio characteristic, if the engine were operating in the subcritical region, the control would sense the lower pressure ratio and continue to increase the fuel flow in an attempt to raise the pressure ratio; the pressure ratio would continue to decrease and the error would be compounded. Therefore, for critical engine operation, this type of control would be unsatisfactory.

Setting diffuser-exit Mach number or scheduling this parameter with flight Mach number overcomes the principal disadvantage of the system operating on diffuser-exit pressure ratio. Figure 5 shows a block diagram of such a system. As in the previous system, flight Mach number is sensed and the signal is fed into a scheduling device, the output of which is the required diffuser-exit Mach number. The measured diffuser-exit Mach number is compared with the required value in the error-detecting device and the difference between these values actuates the fuel control, which varies the fuel flow in such a manner as to reduce the error to zero; the desired operating point is thus attained. This system possesses all the advantages of the previous system, compensates

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for the changes in altitude pressure, and is also unaffected by burner efficiency. Further, it cannot be "fooled" into subcritical operation, as can be deduced from figure 6, which shows the variation of diffuser-exit Mach number with fuel flow at constant flight Mach number. The diffuser-exit Mach number is seen to be a continuously decreasing function of fuel flow; if the engine should get into the subcritical region, the value of the exit Mach number would be too low, which would cause the control to decrease the fuel flow and thus bring the engine back towards its critical point. Of course, this system could produce subcritical operation if the schedule were in error and called for too low a value of diffuser-exit Mach number. The control would maintain subcritical operation, but would not continually increase the fuel flow.

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It should be noted that the control systems just proposed can be used to attain modes of engine operation other than critical-point operation, since at any flight Mach number there exist unique relations between diffuser pressure ratio, diffuser-exit Mach number, and engine thrust. To attain any desired mode of engine operation, within the capabilities of the ram jet, merely requires using a different control schedule relation in the systems presented. The discussion has been limited to critical-point operation because, as previously stated, this manner of engine operation is desirable for long-range missiles.

In order to attain critical operation of the engine, the controls considered to this point have relied upon a schedule of a controlled engine variable with flight Mach number and as such are subject to errors and inaccuracies caused by possible variations in engine characteristics due to manufacturing tolerances and other similar unavoidable factors. The ram-jet engine possesses, however, a property which makes it possible to eliminate such sources of error and to utilize the capabilities of the engine to the fullest extent. This property is, paradoxically, the peak in the pressure ratio curve which was the source of difficulties in the first control system considered. The coincidence of this peak with the point of optimum engine performance makes possible the utilization of the underlying principles of the "optimalizer" control as proposed by Draper and Li of M.I.T. (reference 1). Such a system will seek the point of maximum diffuser pressure ratio at all times and thus will achieve critical engine operation without a schedule.

The principle of operation of this control may be illustrated by a consideration of figure 7. In this figure is reproduced the variation of diffuser pressure ratio P_2/p_0 with fuel flow W_f at constant flight Mach number. The control operates in the following manner: A small periodic variation in W_f is sent into the combustion chamber; this disturbance is superimposed on the mean W_f. Changes in P_2/p_0 result

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from the variation in W_f and this variation is measured and compared with the impressed W_f variation in the following manner: If the engine is in the supercritical region then, as can be seen from figure 7, an increase in W_f results in an increase in P_2/p_0 and for a sinusoidal W_f variation, the variation of these parameters with time will appear as shown in the lower left. The variations of W_f and P_2/p_0 are in phase. This in-phase relation between input and output is detected by a phase-sensitive device. The output signal of the phase detector causes the control to increase the average fuel flow and thereby brings the engine towards the critical operating point. On the other hand, if the engine is in the subcritical region it can be seen that an increase in W_f will result in a decrease in P_2/p_0. This condition is illustrated in the lower right corner of figure 7. In this region, the variation in P_2/p_0 is 180° out of phase with W_f. Since the phase detector senses this condition, the control acts to decrease the mean W_f for this input-output relation and therefore will bring the engine back towards the critical point. This control will seek peak pressure ratio at all times and in this manner satisfy the requirement for critical engine operation. There will, of course, be the continuous variation in thrust F caused by the variation in W_f; this change can be kept small, as determined by the minimum input signal size acceptable for control purposes.

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This system therefore combines the advantages of the scheduling system, that is, it is independent of altitude and combustion efficiency, with the additional advantage that no predetermined schedule or setting is needed to attain critical operation of the engine.

Thus far only the problems involved in attaining a desired engine operating point have been considered. The interrelation between missile-flight-plan requirements and ram-jet performance will now be discussed. During cruise, the missile is required to maintain constant flight speed. The operating map of a ram-jet engine is shown in figure 8, in which is plotted the variation of corrected thrust with flight Mach number at several constant corrected fuel flows. The cruise flight speed for the missile considered is Mach number 3 and at this point engine thrust must equal the missile drag. A typical missile drag curve at constant angle of attack is superimposed on the engine map such that cruise with critical engine operation at Mach number 3 is attained. Critical engine operation exists at the peak of the thrust curves at constant fuel flow. To the right of the peaks in the constant W_f lines is the supercritical region, and to the left is the region of subcritical engine operation.

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The stability of this system is now considered. For an engine operating at constant corrected fuel flow, an external velocity disturbance which increases missile velocity places the missile in a region where the missile drag is greater than the engine thrust; a decelerating force is thereby provided which acts to restore the missile to its cruise velocity, a stabilizing effect and hence desirable. On the other hand, if the disturbance acts to decrease flight velocity, the missile is in a region where drag is greater than thrust and a decelerating force exists which will act to reduce missile velocity still further, and therefore the system is unstable in flight velocity. If an engine control designed to maintain critical engine conditions were in operation, the engine thrust line would be that passing through the peaks in the constant-fuel-flow lines. Under this mode of engine operation, the system is unstable for both increases and decreases in velocity because below cruise velocity the drag would always be greater than thrust and continuous deceleration would occur; above cruise velocity, the thrust would always be greater than the drag and continuous acceleration would result. This velocity instability represents a prime missile-control problem during cruise.

Two means for stabilizing missile velocity are discussed. The first is to utilize available aerodynamic forces by varying the drag of the missile; if, for example, the missile is below cruise speed, the angle of attack may be decreased, which lowers the drag and lift and places the missile in a shallow dive to increase the velocity and vice versa. The second method is to vary engine thrust through fuel flow to provide a restoring force for the missile. For critical engine operation during cruise, this normally cannot be done safely, for an increase in fuel flow results in subcritical operation which may produce engine instability and blow out the burner. It is better to design the engine for supercritical operation at the cruise velocity which then provides, with safety, the necessary latitude of thrust variation to stabilize flight velocity. This, however, requires an oversized engine to propel the missile, which results in an increase in missile weight, because supercritical engine thrust is less than that attainable at the critical point.

In order to determine the nature of missile behavior during transients when the missile is operating under controls utilizing these methods of missile velocity stabilization, the dynamics of an idealized missile-engine combination were investigated with the aid of a differential analyzer for a representative missile configuration. Figure 9 shows a block diagram of a possible control system for cruise velocity stabilization utilizing aerodynamic forces to effect the stabilization. The missile flight Mach number is measured and compared with the required Mach number in an error detector. If the missile is below speed, the difference between measured and set Mach number, the error, acts through

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an angle of attack control to decrease the angle of attack in proportion to the magnitude of the error. This decrease in angle of attack lowers the drag and lift of the missile and places it into a shallow dive. As the missile accelerates, the error decreases and the angle of attack is restored to its equilibrium value. The engine control plays a minor role in this system. Essentially it should be designed to maintain maximum safe thrust during a transient when the missile is below speed and a minimum safe thrust when the missile is above speed. As such, this can be the critical-point operation control discussed previously with a limiting device added for the overspeed condition. For the missile configuration studied, it was assumed that the time required to change the attitude of the missile in pitch was small compared with the time required to change flight speed, and the equations of motion of the missile were solved with these dynamics not considered. These conditions, of course, must be considered in the design of an actual control system.

Figure 10 shows the response of the missile considered to a disturbance in flight velocity under the action of the cruise velocity control of figure 9. Plotted are the variation of velocity and altitude with time for an initial departure from cruise velocity of minus 100 feet per second. The control considered was a simple proportional device with a gain of 3° of angle of attack per 100 feet per second error in flight velocity. This gain merely indicates the magnitude of the corrective action the control takes per unit departure from the desired condition. It is seen that for this type of control action, the missile response is a damped oscillation in both velocity and altitude. The missile passes through the cruise velocity after 1/2 minute but requires about 5 minutes before the oscillation damps to ± 10 feet per second. During the initial oscillation, the altitude of flight drops 4200 feet and after 5 minutes the oscillation in altitude has been damped to ± 400 feet. Increasing the control gain, that is, increasing the change of angle of attack per unit error in flight speed, tends to decrease the response time; this is accomplished, however, by an increase in the excursion in altitude during the first oscillation. The control considered was a pure proportional control and is used merely to illustrate a principle of control rather than to represent optimum control design. A more elaborate system based on the same control principle using derivative or integral action as well as proportional would decrease the oscillatory nature of the missile response and yield shorter response times as well. It should be noted that the response times involved are rather large, but this is typical of aircraft responses at high altitudes because such systems are characterized by high inertia and operate in a rarefied atmosphere with attendant small restoring forces.

The second control system for stabilizing the missile velocity during cruise requires supercritical engine operation. In order to

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investigate this type of stabilization, engine size was increased 19 percent so that the cruise point is as shown on figure 11. For this mode of engine operation the missile is essentially stable in velocity at cruise. For constant fuel flow operation of the engine at velocities greater than cruise velocity, missile drag is greater than thrust and a decelerating force exists which will act to restore the missile to cruise velocity; at flight velocities below cruise but above Mach number 2.8, thrust is greater than drag and an accelerating force exists which would act to restore the missile to cruise velocity. This restoring force is relatively small, however, and the time required to return to equilibrium will be large. In order to decrease the time of response, an engine thrust control may be used to increase the magnitude of the restoring force. A block diagram of such a system is shown in figure 12. The flight Mach number is measured and compared with the required value; if a difference exists, this error acts through the fuel control to increase thrust if the missile is below speed and vice versa, thus producing an increased restoring force to bring the system to its equilibrium point. The use of high control gains for such a system, that is, large changes of fuel flow per unit error in velocity, can result in subcritical operation of the engine during a transient, which is undesirable because of diffuser stability characteristics. In order to prevent such a condition, a limiting device may be incorporated in the control which would prevent subcritical operation. Such a device could be the system considered earlier for attaining critical operation by scheduling diffuser-exit Mach number with flight velocity. This system can be designed to act as a limiter rather than as a primary control and will limit effective fuel flow to safe values. Figure 13 shows the response of the missile acting under such a control scheme, that is, thrust control with limiter action, for an initial displacement from cruise velocity of minus 100 feet per second. It was assumed that the angle of attack was held at a fixed value during the transient by an autopilot. The velocity returns to the cruise value in a more or less exponential manner with a slight superimposed oscillation. About $3\frac{1}{2}$ minutes are required to reach the desired velocity. The missile altitude oscillates during the transient because of the change in lift and settles into a sustained oscillation of small amplitude, about ± 200 feet with a period of approximately 2 minutes. The response time for this particular system is of the same order of magnitude as that of the aerodynamic control system, but it must be remembered that the engine size is 19 percent greater than considered previously and that maximum available restoring force was attained with the limiting control. Therefore, the response time determined may be considered to represent a minimum for this system. For smaller increases in engine size, the available restoring force decreases and larger response times result.

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The choice of cruise control system depends upon many factors. Among these are the flight plan during cruise and the requirements of

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the guidance system. If it is necessary to keep missile weight at a minimum in order to achieve the desired range, then critical engine operation and use of aerodynamic control for velocity stabilization are indicated and the excursions in altitude must be accepted. If weight is not critical, thrust control with supercritical cruise offers the advantages of inherent stability and a minimum of variation in altitude.

The missile guidance system may require a fast response in velocity so that the missile may stay within the limits of a time-programmed flight plan. In this case, aerodynamic stabilization with a high gain control would be required, and excursions in altitude during transient would have to be accepted.

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It can therefore be seen that the selection of a cruise control system must be relegated to the realm of engineering compromise based on the requirements of the particular mission a missile is to fulfill. It is possible, for example, that a combination of the two methods considered would offer such a compromise by combining to some degree the advantages of inherent stability and rapid response.

Thus far control problems for a simple engine with but one variable to manipulate have been considered. The problems associated with a more complicated engine, the turbojet, will now be briefly examined.

As has been shown in previous papers, the turbojet engine has good potentialities as a missile power plant because it can provide sufficient thrust over a range of flight Mach numbers to boost the missile up to its cruise speed and altitude, thus affording appreciable savings in missile gross weight. The desired mode of engine operation is a function of the flight path to be followed during boost and therefore a control system for a boosting engine must be designed with the desired specific path in mind. On the other hand, the basic control loops required for a turbojet engine are, in general, the same since the dynamics of engine operation are not, to any great degree, functions of the flight plan. In order to illustrate the control requirements during boost for a turbojet engine, the mode of turbojet engine operation presented by Mr. Lundin will be considered. This method of engine operation requires the use of a variable-area turbine-inlet nozzle.

A representative boost flight plan for a turbojet missile is shown in figure 14. On this plot are indicated the modes of engine operation required during the various phases of the boost. The missile is designed to take off from a runway; from Mach number 0 to 0.4, full thrust augmentation is required. At Mach number 0.4, augmentation ends and the missile is airborne and accelerates to Mach number 0.8 at constant engine speed N and turbine-inlet temperature T_T; at this flight speed, the

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missile begins a climb to 35,000 feet under the same engine operating conditions. At 35,000 feet, the missile accelerates through the transsonic region at constant altitude until it reaches Mach number 1.2, at which point it goes into a final climb and acceleration to the initial cruise condition. At about Mach number 1.5, the mode of engine operation is changed to constant corrected engine speed with fixed compressor pressure ratio P_3/P_2 and turbine-inlet temperature. It is in this region, from about Mach number 1.5 to the cruise point at Mach number 2.5 at 55,000 feet, that variation of turbine-inlet area is required.

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The variation of engine speed and turbine-inlet area required during the flight plant for this manner of engine operation is shown in figure 15. These variations are expressed as ratios of the value of the variable to its value at take-off. Both speed and area are constant up to about Mach number 1.5. At this point, the speed is required to increase with $\sqrt{\theta}$ and the engine speed at cruise velocity reaches about 130 percent of the take-off value. In this region, the turbine-inlet area A_T must be continuously decreased as flight speed increases and reaches 80 percent of the take-off value at the cruise velocity. During the entire boost and cruise, the turbine-inlet temperature is constant. The compressor pressure ratio is maintained constant during the constant-corrected-speed operating region.

These curves therefore represent control relations to be maintained by manipulation of the engine input variables. It should be emphasized that the schedules shown here are not necessarily the optimum mode of operation for this type of turbojet. The variations shown are merely illustrative and any other schedule within the capabilities of the engine may be used. For this engine, the input variables are primary engine fuel flow $W_{f,E}$, exhaust-nozzle area A_n, afterburner fuel flow $W_{f,AB}$, and turbine-nozzle area A_T. This system therefore has four degrees of freedom, and many control loop configurations may be envisioned to accomplish the desired engine regulation. One such control system that reflects current turbojet control practice is shown in figure 16.

The control consists of a series of scheduling devices, indicated by the boxes labeled S, which receives a signal of missile flight Mach number. The outputs of the schedule are the required values of the engine output variables. These desired magnitudes are attained by manipulation of the engine input variables in closed-loop controls. The engine output (thrust) acts on the missile to change the flight speed, which is sensed and fed into the schedule, closing the loop.

In the system shown, engine speed is controlled by exhaust-nozzle area, turbine-inlet temperature is controlled by primary engine fuel

flow, and afterburner temperature is controlled by afterburner fuel flow. These loops comprise a typical control system for an afterburning turbojet. The fourth loop indicated, which is relatively new to turbojet engine controls, regulates compressor pressure ratio by varying the turbine-nozzle area.

The large number of control loops required for this type of engine present a complicated stability problem. It should be emphasized that the stability problem is not that of the individual loops but is that of the total control with its inherent interaction. By interaction is meant that a disturbance within the engine can cause changes in many of the regulated variables instituting corrective action by the control elements for the affected variables - a situation which can lead to sustained oscillations within the engine and possible engine failure. For example, consider the effect of a disturbance in primary engine fuel flow. This disturbance not only affects turbine-inlet temperature, but it changes engine speed and compressor pressure ratio as well. The change of engine speed is sensed by the speed control, which causes a change in nozzle-exit area, which in turn changes turbine-inlet temperature; the resulting change in fuel flow starts the cycle again. The multiplicity of loops required for such an engine compounds this effect and careful control design is required to prevent instability.

Although this problem exists, it is not insurmountable. There are many control systems of this nature operating satisfactorily for afterburning turbojets which have three degrees of freedom; the fourth loop merely adds to the design problem. Methods for stabilizing multiloop turbojet engine controls are discussed in detail in reference 2.

At the cruise condition, turbojet-propelled missiles can have the same sort of flight velocity stability problem discussed previously for the ram jet. The same stabilization considerations apply to the turbojet; that is, either aerodynamic force or thrust control may be used to stabilize velocity. The method selected is again dependent upon the requirements posed by the specific mission involved.

Some of the control problems associated with air-consuming propulsion systems for long-range missiles have been discussed in this paper. Control systems for attaining desired modes of engine operation and for stabilizing flight speed have been suggested and their advantages and disadvantages discussed. The systems and methods presented represent principles of control methods rather than optimum control designs and, as previously stated, any choice of a control system must be an engineering compromise based on the requirements of a specific missile and mission.

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1. Draper, C. S., and Li, Y. T.: Principles of Optimalizing Control Systems and an Application to the Internal Combustion Engine. Aero. Eng. Dept., M.I.T., Sept. 1951.

2. Boksenbom, Aaron S., and Hood, Richard: General Algebraic Method Applied to Control Analysis of Complex Engine Types. NACA Rep. 980, 1950. (Supersedes NACA TN 1908.)

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REQUIRED VARIATION OF FUEL FLOW FOR CRITICAL ENGINE OPERATION

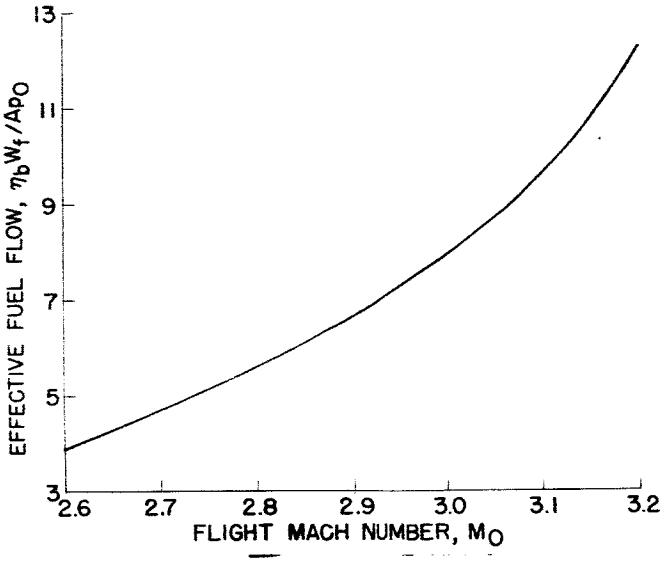

Figure 1

REQUIRED VARIATION OF DIFFUSER PRESSURE RATIO AND EXIT MACH NUMBER FOR CRITICAL ENGINE OPERATION

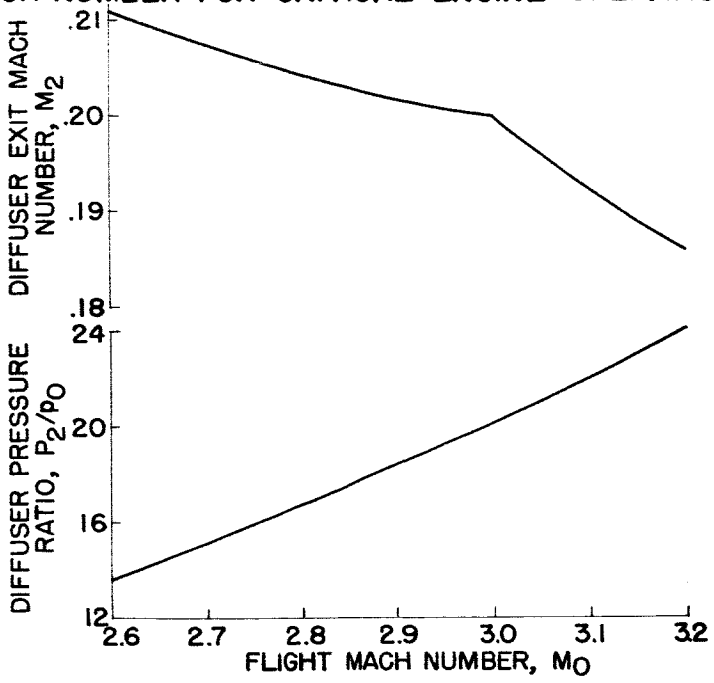

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Figure 2

ENGINE CONTROL WITH P_2/p_0 SCHEDULED WITH FLIGHT MACH NUMBER

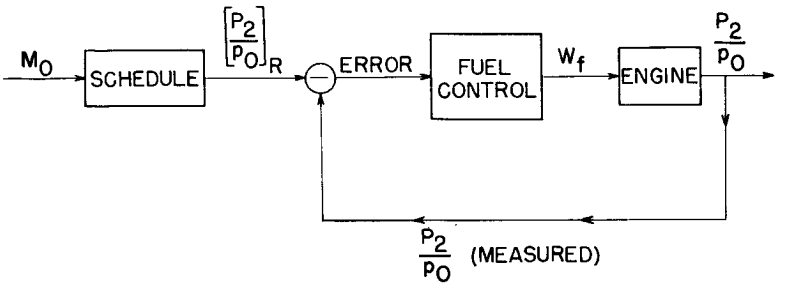

Figure 3

EFFECT OF FUEL FLOW ON DIFFUSER PRESSURE RATIO AT CONSTANT FLIGHT MACH NUMBER

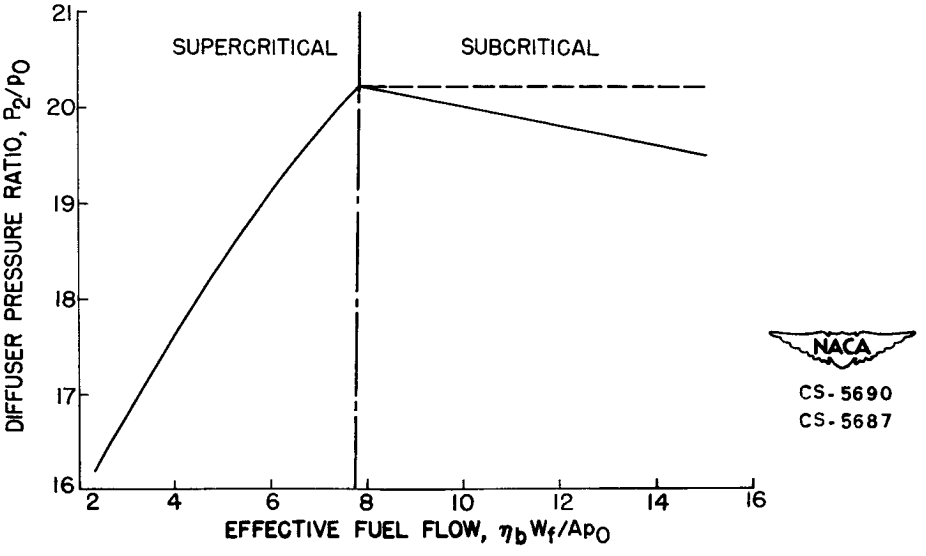

Figure 4

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ENGINE CONTROL WITH M_2 SCHEDULED WITH FLIGHT MACH NUMBER

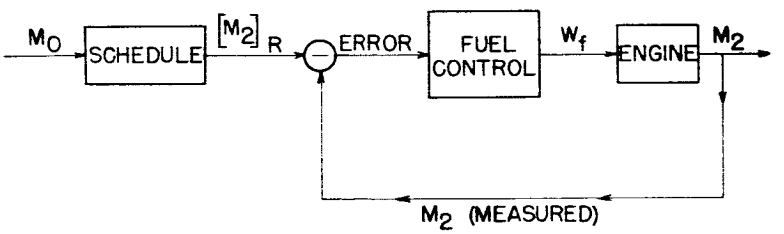

Figure 5

EFFECT OF FUEL FLOW ON DIFFUSER EXIT MACH NUMBER AT CONSTANT FLIGHT MACH NUMBER

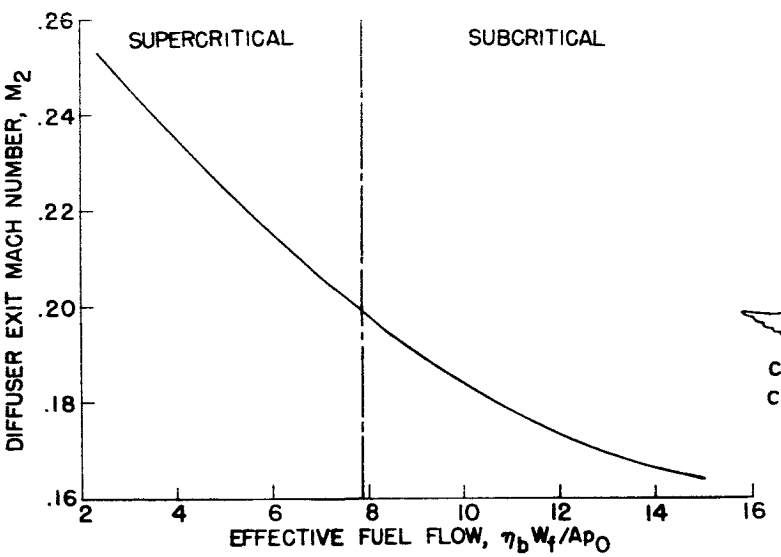

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Figure 6

PRINCIPLE OF OPERATION OF OPTIMIZING CONTROL SYSTEM

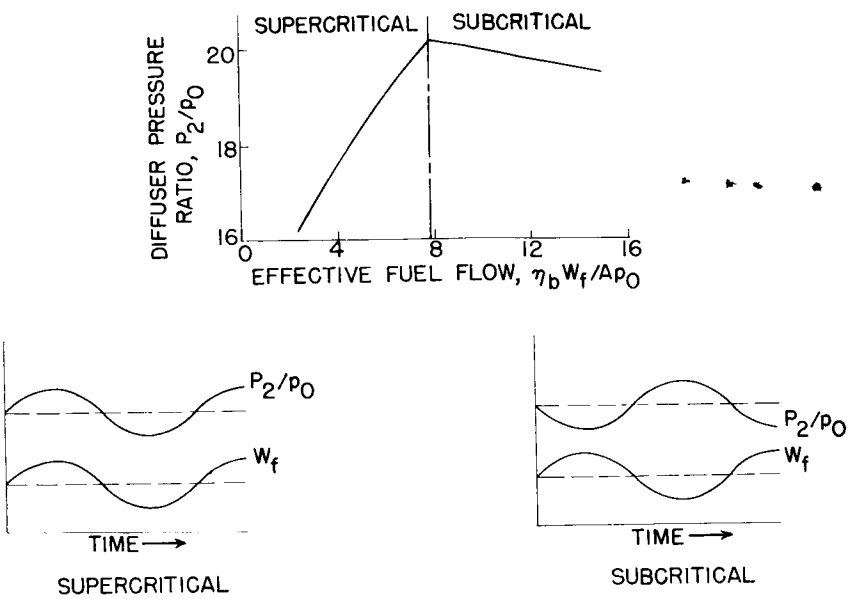

Figure 7

RAM-JET PERFORMANCE FOR CRITICAL CRUISE

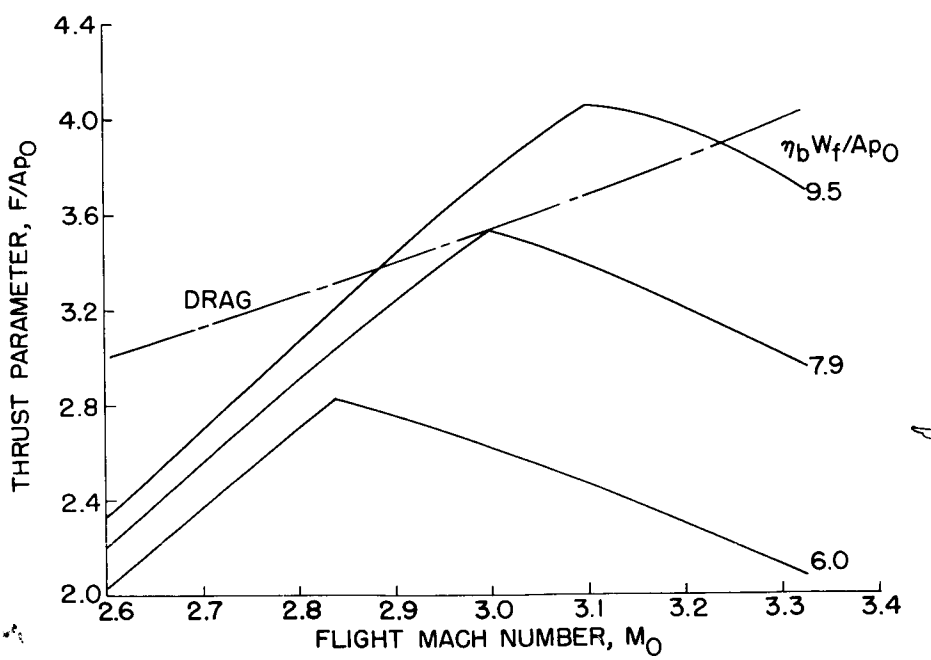

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Figure 8

[redacted]

CRUISE CONTROL OF MACH NUMBER THROUGH ANGLE OF ATTACK

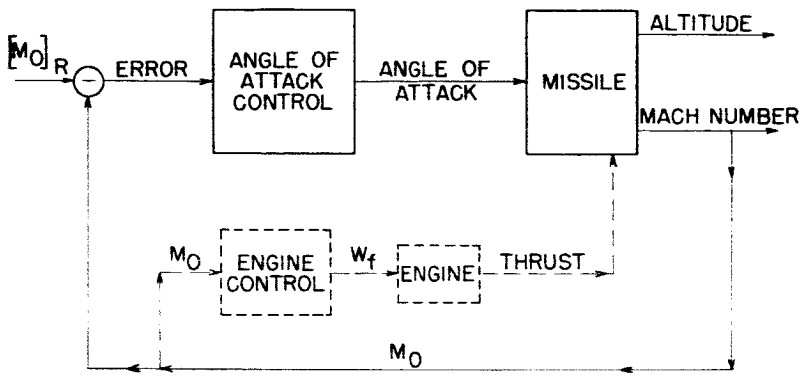

Figure 9

RESPONSE OF MISSILE TO VELOCITY DISTURBANCE WITH AERODYNAMIC VELOCITY STABILIZATION AND CRITICAL ENGINE

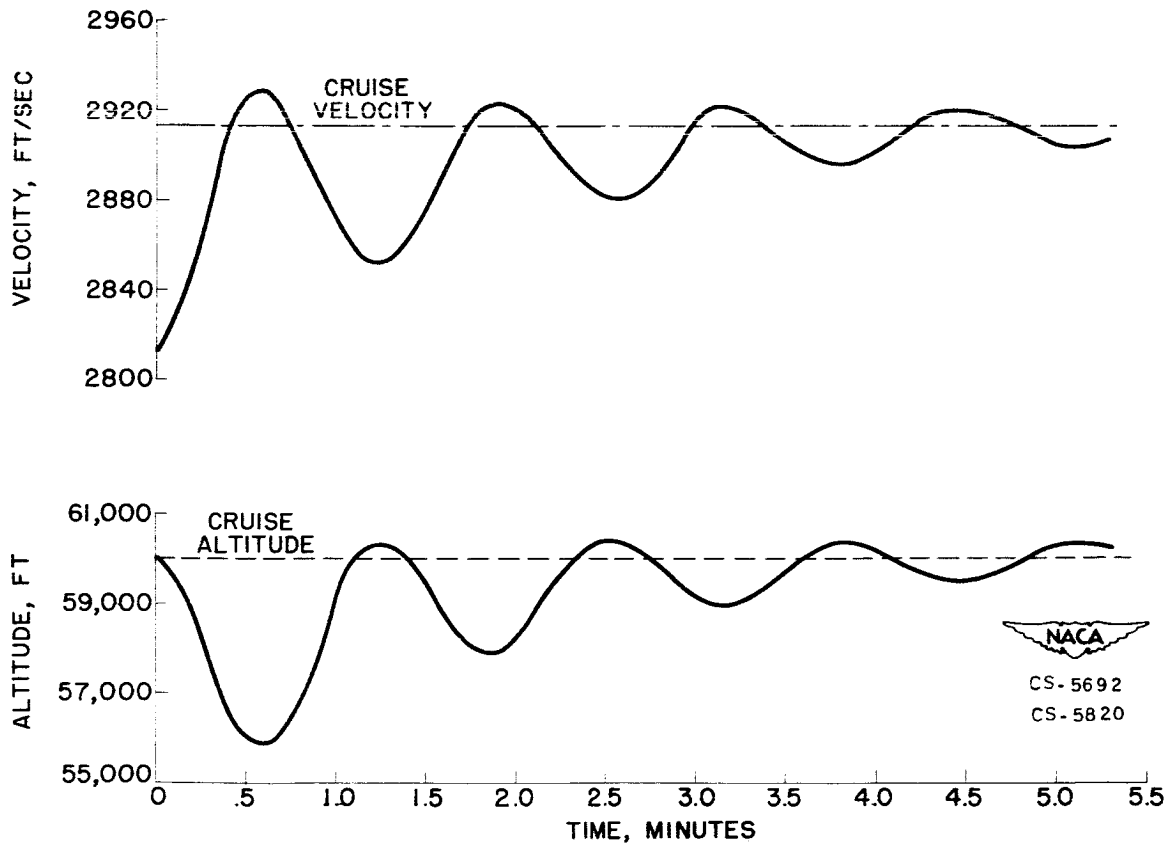

Figure 10

[redacted]

[redacted]

RAM-JET PERFORMANCE FOR SUPERCRITICAL CRUISE

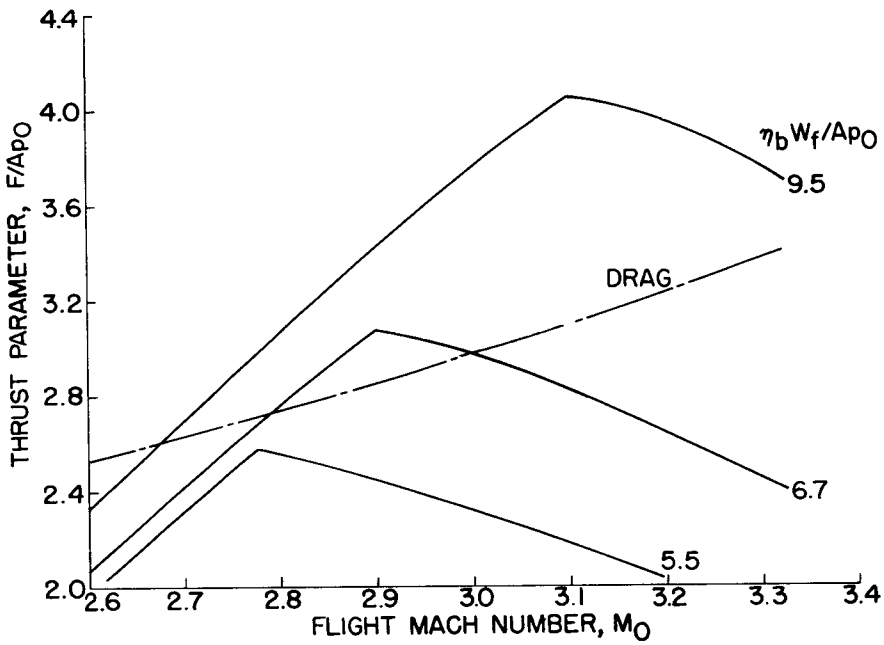

Figure 11

CRUISE CONTROL OF MACH NUMBER THROUGH ENGINE THRUST

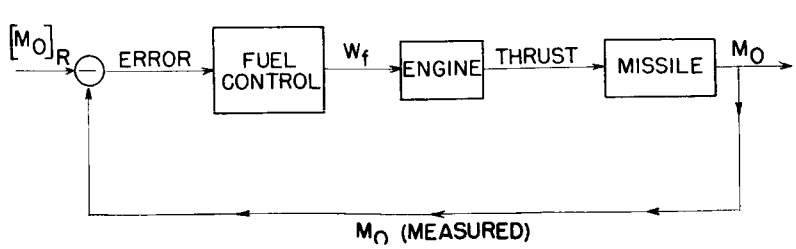

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Figure 12

[redacted]

RESPONSE OF MISSILE TO VELOCITY DISTURBANCE WITH THRUST CRUISE VELOCITY CONTROL

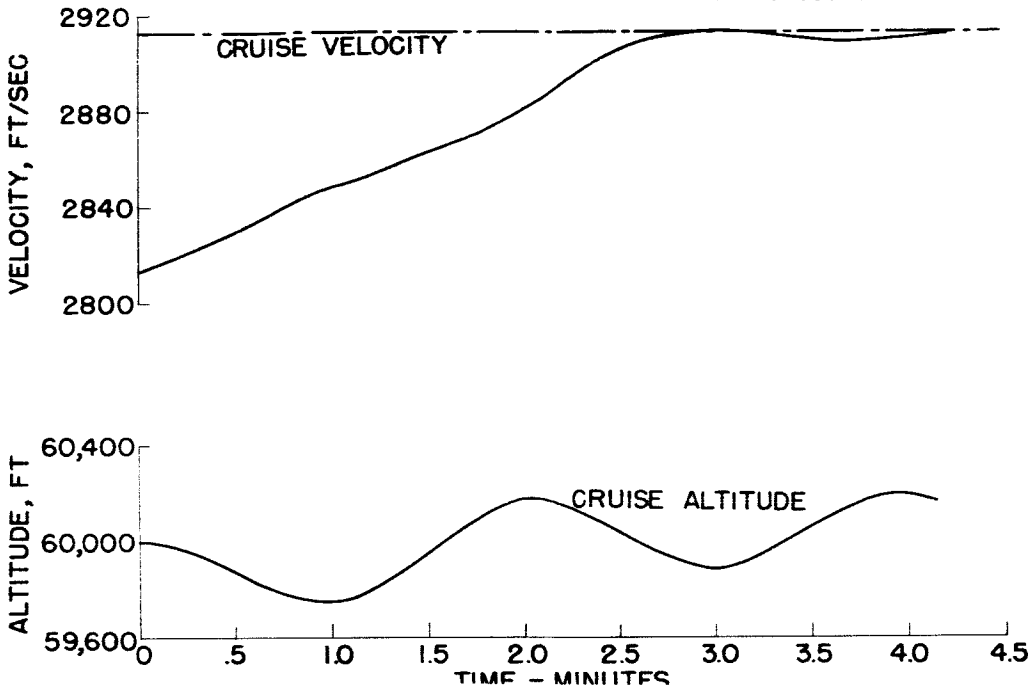

Figure 13

BOOST FLIGHT PATH FOR TURBOJET MISSILE

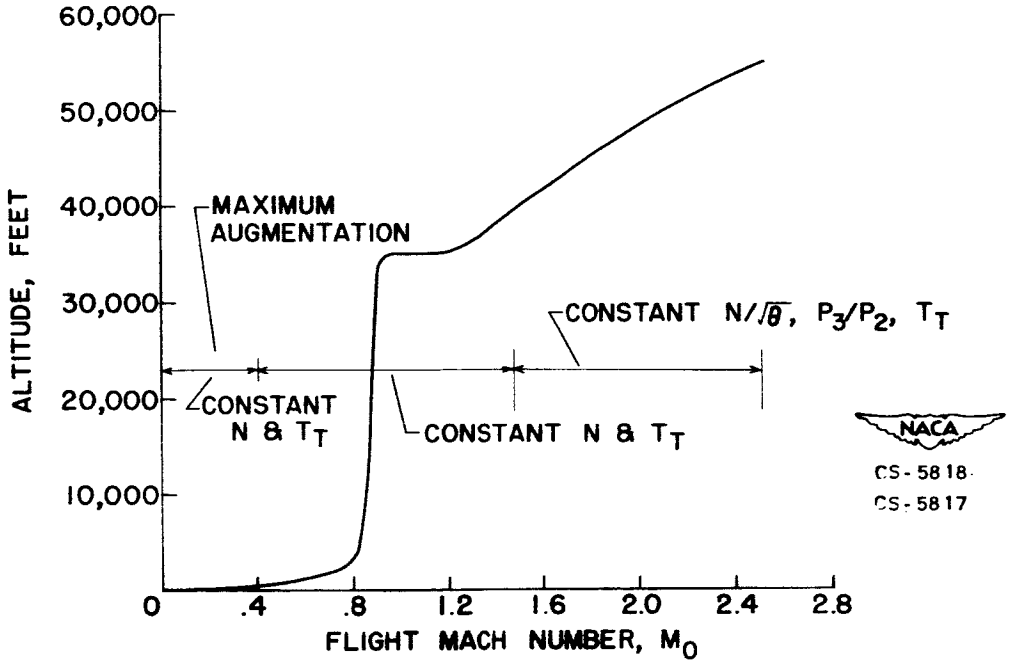

Figure 14

REQUIRED VARIATION OF TURBOJET ENGINE PARAMETERS DURING BOOST

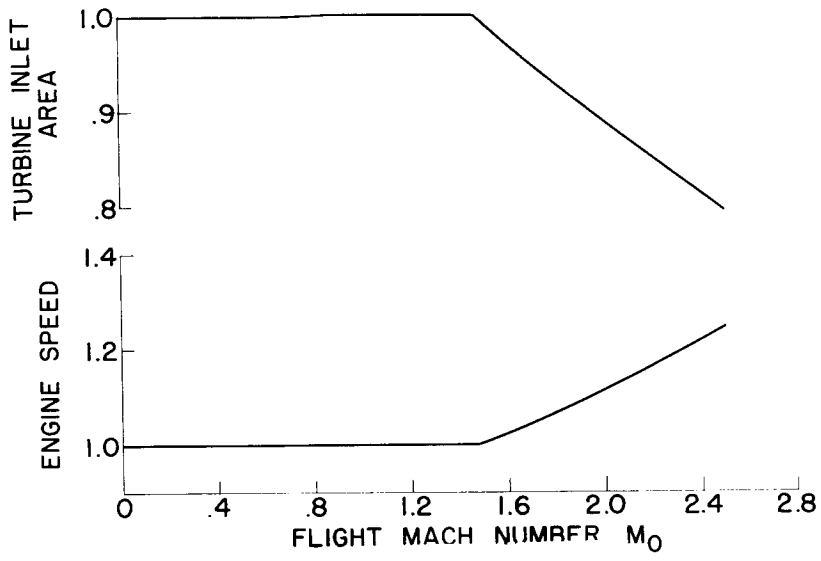

Figure 15

CONTROL SYSTEM FOR TURBOJET ENGINE

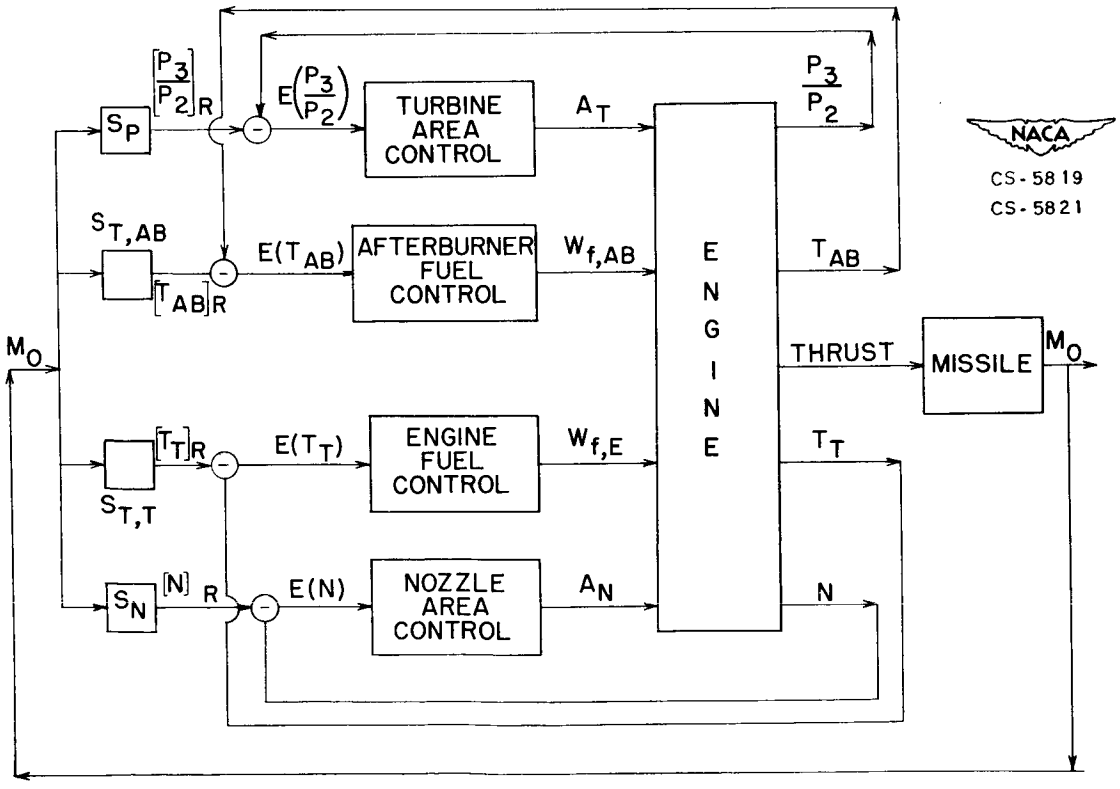

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Figure 16